NASA Conference Publication 3112, Vol. 2

Space Transportation Propulsion Technology Symposium

Volume 2—Symposium Proceedings

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Proceedings of a symposium held at
the Pennsylvania State University
State College, Pennsylvania
June 25–29, 1990

NASA

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Space Transportation Propulsion Technology Symposium

Volume 2—Symposium Proceedings

NASA Office of Space Flight
Washington, D.C.

Proceedings of a symposium held at
the Pennsylvania State University
State College, Pennsylvania
June 25–29, 1990

NASA
National Aeronautics and
Space Administration
Office of Management
Scientific and Technical
Information Program
1991

THE SPACE EXPLORATION INITIATIVE

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DEVELOPMENT OF SYMPOSIUM THEMES

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PLENARY SESSION

3:15-3:30	*Break (Beverages available)*- Lobby, Kern Graduate Center & 2nd floor, Willard Building	
5:30-6:00	Resolution of Issues (If Required)	Panel Leaders & Staff
6:00-7:00	*Social Mixer*- Lobby, Days Inn	PSU Staff
7:00-8:30	*Banquet*- Banquet Room, Days Inn Speaker: Mr. James McDivitt Senior Vice President Rockwell International	All

Thursday, 28 June

7:00-8:00	*Breakfast: Waring Commons* (Registration Continues- Lobby, Kern Graduate Center)	PSU Staff

BREAKOUT SESSIONS

8:00-2:00	**PANELS RECONVENE**- Various rooms in Willard Building Focus: Document Findings, Summarize, Prepare Briefings. Note: Computer Chart Making Support Available in 101A, Kern Graduate Center	Panel Leaders and Members
10:00-10:15	*Break (Beverages available)*- Lobby, Kern Graduate Center & 2nd floor, Willard Building	
12:00-1:00	*Luncheon:Waring Commons*	PSU Staff

PLENARY SESSION

2:00-5:30	NASA Propulsion Engineering Research Center at Penn State, Second Annual Symposium- Concurrent sessions in rooms 101 and 112, Kern Graduate Center (See enclosed agenda)	PSU Staff
(As Avail/Req'd)	Rapporteur's Perceptions and Critique of Panel Deliberations and Results	Council of Rapporteuers (Off Line to Staff)
3:30-3:45	*Break (Beverages available)*- Lobby, Kern	
6:00-7:30	*Picnic*- Lawn of Hetzel Union Building (Inside HUB if inclement weather)	PSU Staff

Friday, 29 June

7:00-8:00	*Breakfast: Waring Commons*	PSU Staff
8:15-9:00	*Speaker:* The Honorable Robert S. Walker, U.S. House of Representatives	All
9:00-9:30	Panel A Reports (to Plenary Session)	Panel A
9:30-10:00	Panel B Reports (to Plenary Session)	Panel B
10:00-10:15	*Break (Beverages available)*- Lobby, Kern Graduate Center	
10:15-10:45	Panel C Reports (to Plenary Session)	Panel C
10:45-11:15	Panel D Reports (to Plenary Session)	Panel D
11:00-12:00	Open Discussion, Summary of Conclusions and Closing Remarks (Revew of Findings, etc.)	R. Schwinghamer, C. Vaughan, W. Wiley
12:00-1:00	*Luncheon: Waring Commons/Symposium Adjournment*	

4:10-5:10	Booster Propulsion - Liquids/Hybrids Solids	U. Heuter, MSFC C. Clinton, MSFC R. Lund, Thiokol
5:10-5:30	ALS	J. Monk, MSFC
5:30-5:50	**ENVIRONMENTAL CONSIDERATIONS**	J. Jatko, NASA HQI
6:00-7:30	NASA Propulsion Engineering Research Center at Penn State- Facilities tour followed by: *Social Mixer: Wine & Cheese (Shuttle Buses will operate between Kern and Center facilities)* *Dinner on your own*	PSU Staff

Wednesday, 27 June

7:00-7:50	*Breakfast: Waring Commons* (Registration Continues- Lobby, Kern Graduate Center)	PSU Staff
	PLENARY SESSION- 112 Kern Graduate Center	
7:50-8:00	Announcements	
	NEXT GENERATION - Input to Panels (Cont'd)	
8:00-8:20	AF Space Systems Propulsion	D. Hite, AFAL
8:20-8:40	Unmanned Launch Vehicles/Upper Stages	C. Gunn, NASA HQ
8:40-9:20	Space Transfer Vehicles	F. Huffaker, MSFC B. Tabata, LeRC
9:20-9:40	Advanced Manned Launch Systems (AMLS)	D. Freeman, LaRC
9:40-10:00	National Aerospace Plane (NASP)	M. Tang, NASA HQ
10:00-10:20	*Break (Beverages available)*- Lobby, Kern Graduate Center	
10:20-11:20	**FOREIGN TECHNOLOGY** - Input to Panels	
	- Japanese Technology	C. Merkle, Penn State
	- Russian Technology	R. Jones, Rocketdyne
	- European, Other Technology	E. Rice, Orbitec
11:20-12:40	**FUTURISTIC SYSTEMS** - Input to Panels	
	- Nuclear and Solar Electric Propulsion	D. Byers, LeRC
	- Nuclear Thermal Propulsion	G. Bennett, NASA HQ
	- Fusion Propulsion	N. Schulze, NASA HQ
	- Advanced Propulsion Concepts	R. Frisbee, JPL
12:40-1:40	*Luncheon: Waring Commons*	PSU Staff
	BREAKOUT SESSIONS	
1:40-5:30	**PANELS CONVENE**- Various rooms, Willard Building (See enclosed map) Note: Computer chart making support available - 101A, Kern Graduate Center	Panel Leaders and Members

AGENDA

SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM

The Pennsylvania State University, University Park, PA 25-29 June 1990

Monday, 25 June

4:00-8:00	Registration: Badge, Agenda (final), Preprints, Banquet ticket, Visitor info, etc. (Coffee available)-Lobby, Nittany Lion Inn	PSU Staff
5:00-6:30	*Social Mixer* - Ticketed Participants & Guests- Colonial Room, Nittany Lion Inn	PSU Staff
6:30-8:00	*Dinner -Open Evening*	All

Tuesday, 26 June

7:00-8:00	*Breakfast: Waring Commons* (Registration Continues- Lobby, Kern Graduate Center)	PSU Staff

PLENARY SESSION- 112 Kern Graduate Center

8:00-8:15	Welcome and Announcements	R. Jacobs, PSU
8:15-9:00	Symposium Overview	
	-Call to Order, General Chairman's Remarks	R. Schwinghamer
	-Co-Chairmen's Comments	C. Vaughan, W. Wiley
	-Headquarter's Perspectives	D. Branscome
9:00-9:45	Keynote Address- James R. Thompson, Jr. NASA Deputy Administrator	All
9:45-10:00	*Break (Beverages available)*- Lobby, Kern Graduate Center	PSU Staff
10:00-12:30	Development of Symposium Themes	
	-Space Exploration Initiative	C.C. Priest, NASA HQ
	-National Space Transportation Strategy	D. Branscome, NASA HQ
	-Maintaining Technical Excellence	T. Davidson, AIA
	-Operational Efficiency - New Approaches to Future Propulsion Systems	R. Rhodes, KSC G. Wong, Rocketdyne
12:30-1:30	*Luncheon: Waring Commons*	PSU Staff

PROPULSION SYSTEM OPTIONS: Systems/Requirements Input to Panels

	CURRENT SYSTEMS - Input to Panels	
1:30-1:50	Expendable Launch Vehicle Propulsion	P. Fuller, Rocketdyne
1:50-2:10	Shuttle Propulsion Systems	R. Bardos, NASA HQ
2:10-2:50	Upper Stages/Propulsion	C. Gunn, NASA HQ J. Brown, P&W
2:50-3:10	Satellite/Spacecraft Propulsion	M. Dowdy, JPL
3:10-3:30	*Break (Beverages available)*- Lobby, Kern Graduate Center	
	NEXT GENERATION - Input to Panels	
3:30-4:10	Shuttle Derivatives - Manned Unmanned	W. Ordway, JSC U. Heuter, MSFC

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INTRODUCTION

The Space Transportation Propulsion Technology Symposium (STPTS) was held at the Pennsylvania State University in University Park, PA, June 25-29, 1990. The Symposium consisted of a two-day plenary session, a one-day breakout session for the meeting of four individual panels, and a concluding morning session for the presentation of panel summary reports. In addition to the Symposium, the Second Annual Symposium of the NASA Propulsion Engineering Research Center at Penn State was held concurrently on the third day.

The STPTS Executive Summary, NASA Conference Publication 3112 Volume 1, contains the conclusions and recommendations of the Symposium participants as well as a description of the Symposium activities. The Symposium proceedings are organized in five sections and are contained in NASA Conference Publication 3112 Volumes 2 and 3.

This document, Volume 2 of NASA Conference Publication 3112, includes Section 1, the plenary session presentations, and Section 2, the Second Annual Symposium of the NASA Propulsion Engineering Research Center at Penn State.

Volume 3 of NASA Conference Publication 3112 contains the remainder of the STPTS proceedings. Section 3 contains the panel summary reports, Section 4 contains the papers and briefing materials presented to the four panels, and Section 5 contains the list of STPTS participants. Volumes 2 and 3 also contain the STPTS agenda, a description of the topics discussed by the four panels, and the table of contents for the other volume in the appendix.

SPACE TRANSPORTATION PROPULSION SYSTEMS SYMPOSIUM

Panel Topics

GENERAL CHAIRMAN
ROBERT SCHWINGHAMER

COCHAIRMEN
CHESTER VAUGHAN - JSC, WARREN WILEY - KSC

PROPULSION SYSTEM OPTIONS

CURRENT SYSTEMS
- o ELVs - Small, Med, Lrge
- o Shuttle - SSME, OMS, RCS, RSRM, ASRM
- o Upper Stages
- o Satellite/Space Probe Prop

NEXT GENERATION (Candidates)
- o Shuttle Derivatives
- o Booster Propulsion (Liquid, Hybrid, Solid
- o Advanced Launch Systems
- o Unmanned Launch Vehicles and Upper Stages
- o Space Transfer Vehicles/Adv. Cryo. Propulsion
- o Adv. Manned Launch Sys. (Shuttle II, SSTO, Comb. Cycle Prop., Adv. Rockets)
- o NASP

ENVIRONMENTAL ISSUES

FOREIGN TECH.
- o Japanese o Russian
- o European o Other

FUTURISTIC SYSTEMS (Candidates)
- o Nuclear Thermal Propulsion (Fission, Fusion)
- o Nuclear &Solar Electric
- o Adv. Propulsion Concepts

* Panel Rapporteur/Facillitator

SYSTEMS ENGINEERING & INTEGRATION

+ Len Worlund - MSFC
Phil Deans - JSC
Frank Berkopec - LeRC
(+ Panel Leader)

- **o Prelim Design Activities**
 - ° Conceptual Design (Phase A Studies)
 - ° Pre Development/ Phase B Studies
 - ° System Architecture
 - ° Vehicle End-to-End Subsystem Interdependencies
 - ° Trajectory/Performance Planning Options
- **o Phase C/D Activities**
 - ° Pre Development Technology Maturity
 - ° PDR Penetration
 - ° Modular vs LRU's
 - ° FMEA/CIL
 - ° Design Margin
- **o Flight System Evolution**
 - ° Uprating (Perf/Life)
 - ° Cost Reduction
 - ° Assured Access

* Irving Davids
* Carl Aukerman

DEVELOPMENT, MANUFACTURING & CERTIFICATION

+ Walt Karakulko - JSC
Paul Shuerer - MSFC
Steve Dick - SSC

- **o System Development**
 - ° Probabilistic Structural Analysis Methods
 - ° Tech Trfr Methodology
 - ° National Test Bed Concept
 - ° Historical Problem Areas - Solutions Needed
- **o Mat'ls & Manufacturing**
 - ° Manu. Processes & Applications
 - ° National Mat'ls Data Base
 - ° NDE
 - ° Concurrent Engineering
- **o Flight Certification**
 - ° Integration of Diagnostics into Test Process
 - ° Life Cycle Cost Based Test Program Decisions
 - ° Certification Test Requirements - Manrating
 - ° Testing vs. Simulation

* Mel Bryant
* Bill Hope

OPERATIONAL EFFICIENCY

+ Don Nelson - JSC
Russ Rhodes - KSC
Marv Carpenter - SSC
Fred Huffaker - MSFC

- **o Pre-Launch Activities**
 - ° Operationally Efficient Propulsion Systems
 - ° Facilities Requirements
- **o Flight Operations**
 - ° Data Acquisition
 - ° Flight Control
 - ° Weather Limitations/ All Weather Capability
- **o Mission Success Assurance**
 - ° Safety & Diagnostics
 - ° Configuration Control
- **o Space Basing**
 - ° System Concepts
 - ° Propellant Storage/Trfr
- **o Review Survey**
- **o Subpanel Discussions on Ops Efficiency for:**
 - ° Shuttle Derivatives
 - ° ELV's
 - ° Upper Stages/Manned
 - ° Satellites/Deep Space Probes

* Bill Dickinson
* Brenda Wilson

PROGRAM DEVELOPMENT & CULTURAL ISSUES

+ Ed Gabris - HQS
Chuck Eldred - LaRC
Harry Erwin - JSC
Gene Austin - MSFC

- **o Lessons Learned (Shortcomings)**
- **o Requirements**
 - ° Shuttle Level II
 - ° Fixed Capability
 - ° ALS
 - ° Environmental Consid/TQM
 - ° Assured Access to Space
- **o Technology/Perf/Ops**
 - ° Tech Limited
 - ° Perf Driven
 - ° Labor Intensive
 - ° Perf Margin
 - ° Cost Driven
 - ° Skeleton Crews
- **o Reliability/Safety**
 - ° By Test
 - ° Redundancy
 - ° Engine On/Off/Out
 - ° Constraints (Redlines)
 - ° Margin/Design
 - ° Fault Tolerant
 - ° Safety
 - ° Hlth Monitoring
- **o Procurement/Contracting**
 - ° Competitive
 - ° Mission Need Statement/A109
 - ° Yr-to-Yr Funding
 - ° IR&D
 - ° Consortium
 - ° Multi-Yr Funding
 - ° Joint Funding
 - ° IR&D

* Rodney Johnson
* Diane Gentry

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The Space Exploration Initiative

Briefing to

Space Transportation Propulsion Technology Symposium
Pennsylvania State University

June 26, 1990

Pete Priest
Marshall Space Flight Center

NASA

WHY ARE WE GOING TO MARS?

To strengthen our country's international competitiveness
- technology
- education

To continue America's journey into space

To understand planetary evolution

To enhance our understanding of life in the universe and find out if life once existed on Mars

To fulfill the human imperative to explore

Carry out the National Space Policy goal of expanding human presence and activity beyond Earth orbit into the solar system

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WHY GO TO THE MOON FIRST?

Learn to build, live and work on planetary surface close to home

Nearby — a 3-day trip and near instantaneous communications

Human experience in partial gravity leads to Mars

New science opportunities

Significant achievement by early next century

An evolutionary approach to "expanding human presence and activity"

Office of Aeronautics, Exploration and Technology

NASA

PRESIDENTIAL DECISION ON THE SPACE EXPLORATION INITIATIVE

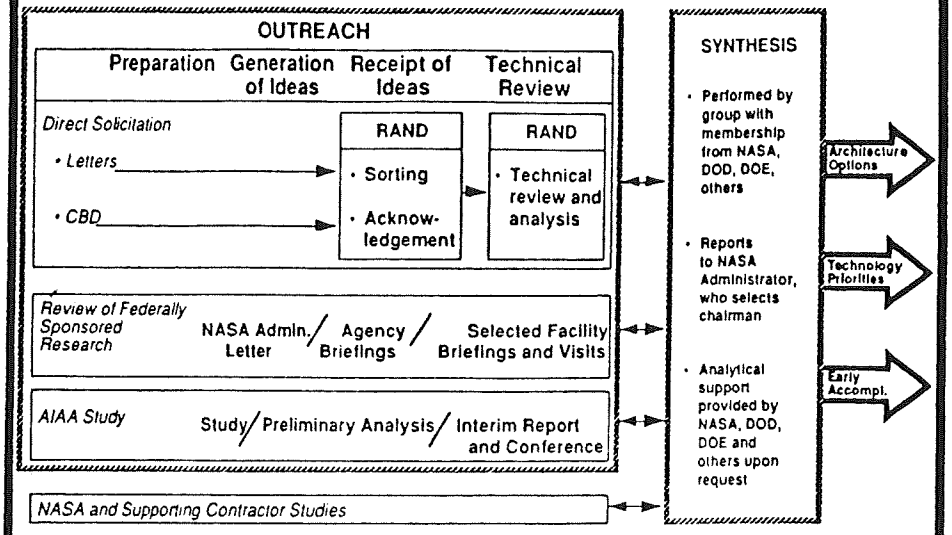
On February 16, 1990 President Bush approved policy for the Space Exploration Initiative:

- Initiative will include both Lunar and Mars program elements, as well as robotic science missions
- Near-term focus will be on technology development
 - Search for new/innovative approaches and technology
 - Investment in high leverage innovative technologies with potential to make a major impact on cost, schedule, and/or performance
 - In parallel with mission, concept, and system analysis studies
- Selection of a baseline program architecture will occur after several years of defining two or more reference architectures while developing and demonstrating broad technologies
- NASA will be the principal implementing agency while DOD and DOE also will have major roles in technology development and concept definition. The National Space Council will coordinate the development of an implementation strategy by the three agencies

Office of Aeronautics, Exploration and Technology

NASA

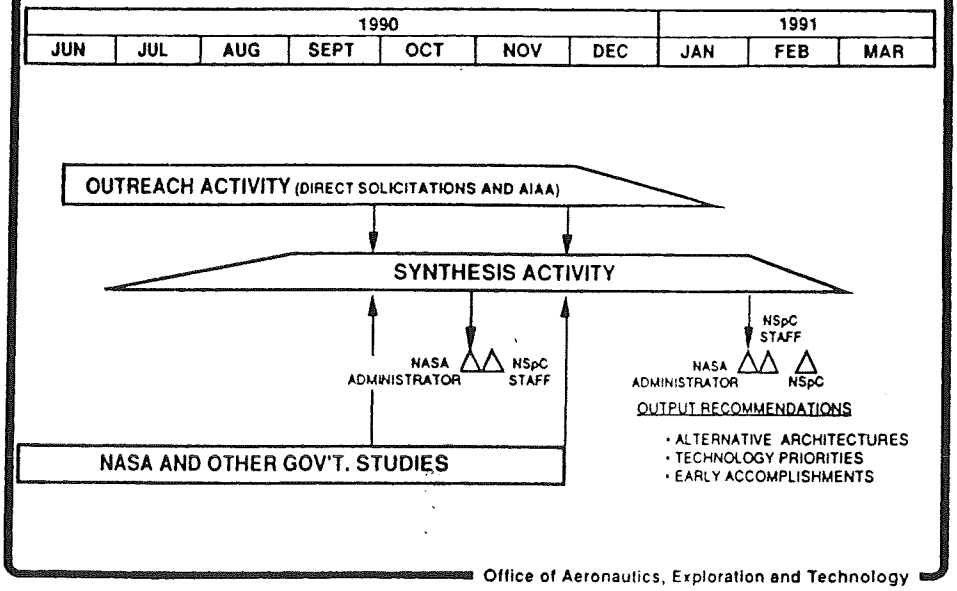
OUTREACH AND SYNTHESIS PROCESS



Office of Aeronautics, Exploration and Technology

NASA

OUTREACH AND SYNTHESIS SCHEDULE



Office of Aeronautics, Exploration and Technology

NASA

LUNAR TRANSPORTATION SYSTEM REQUIREMENT DRIVERS

- **Mass delivered to lunar surface**
 - Crew size
 - Lunar base elements
 - Separate or combined crew/cargo flights
- **Type of lunar base**
 - Support a permanent base
 - Man-tended missions only
 - Evolution of lunar base/date of first mission
- **Design approach**
 - Commonality of cargo/crew vehicles
 - Commonality with Mars transportation system
 - Extent of transportation system reuse
 - Extent of on-orbit operations
 - -- Launch vehicle size
 - -- Expendable versus space-based reusable vehicles

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NASA

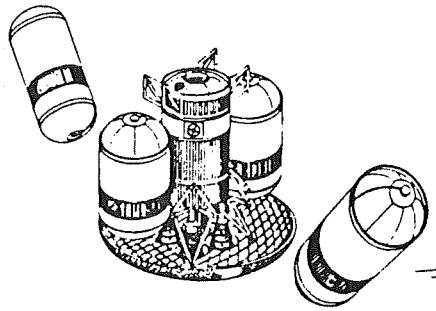
SPACE TRANSPORTATION SYSTEM KEY REQUIREMENTS FOR LUNAR BASE SUPPORT

- Deliver up to 30t of cargo to lunar surface on a single mission
- Deliver 4 crew and up to 15t cargo to lunar surface and return the crew to Space Station Freedom
 - Support continuous human presence at base by crew exchange
 - Support a human tended base by crew sorties to the Moon
- Provide common vehicle design for both cargo and crew delivery to reduce number of hardware developments
- Provide vehicle reuse to reduce vehicle and operational cost
- Use Space Station Freedom as an orbital transportation node for vehicle assembly and staging
- Provide heavy-lift launch vehicle capability that reduces number of launches and on-orbit assembly requirements
 - 60-70t minimum payload to Freedom
 - 7.6 meter payload shroud
- Space transportation system to be available within 10 years

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LUNAR TRANSPORTATION SYSTEM

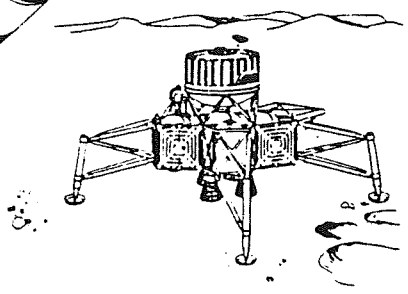


Lunar Transfer Vehicle (LTV)

- Core stage with drop tank design
- Liquid hydrogen/liquid oxygen propellant
- 4 engines at 20K thrust each
- Vehicle mass: 128t

Lunar Excursion Vehicle (LEV)

- Payload to surface: 15t plus crew module
- Single stage design
- Liquid hydrogen/liquid oxygen propellant
- 4 engines at 20K thrust each
- Vehicle mass: 32t



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LUNAR TRANSFER SYSTEM CONCEPT IMPROVEMENTS

- **Reduce number of vehicle elements**
 - Single crew module
 - Single P/A module
 - Fewer propellant tanks
 - Fewer engines
- **Avoid LLO cargo transfer**
 - Mate P/L to lander at SSF
 - Fly cargo mission direct from ETO
- **Avoid LLO propellant transfer**
 - Store return propellant in separate tanks
 - Direct return from lunar surface
- **Avoid engine doors in aerobrake**
 - Locate aerobrake on opposite end from engines
 - Allow smaller penetrations (feedlines: STS proven)
- **Enhance crew module access/visibility**
 - Fewer vehicle elements
 - Configuration rearrangement
- **Minimize assembly at SSF**
 - Direct TO LS cargo flights
 - Reduce number of elements requiring assembly.
- **Improve cargo accommodations**
 - Fly expendable cargo missions
 - Reduce or eliminate cargo on pilot flights
 - Avoid cargo transfer operations (LLO)

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MARS TRANSPORTATION SYSTEM REQUIREMENT DRIVERS

- **Mass delivered to Mars surface**
 - Crew size
 - Mars base elements
 - Separate or combined crew/cargo flights
- **Long duration of the Mars mission**
 - Launch date/trajectory considerations
 - Habitat module impact
 - Need for artificial gravity
 - Need for radiation shielding protection
 - Desire to reduce mission duration
- **Mars aerobraking**
 - Chemical propulsion/aerobrake versus advanced propulsion concepts (NTR, SEP, NEP, GCR)
 - Aerobraking needed for Mars landing from orbit

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KEY REQUIREMENTS MARS TRANSPORTATION SYSTEM

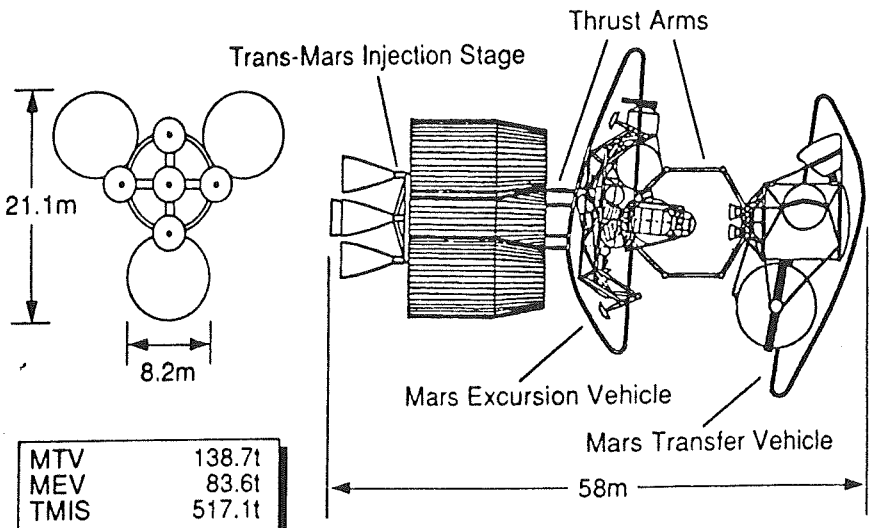
- Deliver 4 crew and 25t cargo to Mars surface on first human landing in 2016
 - Crew and 1t payload returned to LEO
- Deliver 100t of cargo to Mars surface on first cargo flight in 2025
- Provide for reuse (up to two missions) of piloted Mars Transfer Vehicle (MTV)
 - Cargo vehicles and landers are expended at Mars

 Piloted missions utilize zero-g for transit phases of missions
- Chemical propulsion (LOX/LH_2) utilized for all propulsive maneuvers (TMI, TEI, etc.)
- Aerobraking utilized at Mars and Earth arrival
- Provide heavy-lift launch capability that reduces number of launches and on-orbit assembly requirements
 - 140t minimum payload to LEO
 - 13.7 payload shroud

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FULL-UP MARS MISSION VEHICLE IN LEO



MTV	138.7t
MEV	83.6t
TMIS	517.1t
Total IMLEO	739.4t

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MARS TRANSPORTATION SYSTEM CONCEPT IMPROVEMENTS

- **Advanced Propulsion Systems**
 - NTR, GCR, NEP, SEP
 - Parametrics for candidate systems
 - Sensitivities and trade assessments
 - Conceptual Designs
 - Operations and Safety
 - Programmatics
- **Artificial Gravity**
 - "Artificial-g Delta Study" to assess weight, technology, cost and operations penalties
 - Define "From the Start" concept
- **MTS Aerobrake Issues**
 - Aeroheating for Mars aerocapture/entry and Earth aerocapture
 - Landing criteria for cross range, altitude and avoidance maneuvers
 - G&C capture/entry at Mars and Earth
- **MTS Equipment Life and Self Check**
 - Requirements and technical approach to assuring critical equipment operability
 - Parametric examination of sparing level, commonality, spares quantity and MTBF
- **LTS/MTS Crew Modules**
 - Compare MTV space habitat concepts
 - Define common family of habitats for LTV, LEV, MEV and other uses
 - Assess evolutionary growth potential
- **MTS Mission Scenario**
 - Mission analyses of reference and alternate opportunities and profiles from 2009 through 2025
 - Operations sequence assessment for reference system
 - -- Same for reference systems with artificial-g
 - -- Same for advanced propulsion

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Excerpts from "The Exploration Initiative," an additional paper provided by C. C. Priest with his presentation.

This "Exploration Initiative" package is a compilation of selected NASA policy and presentational material on the President's commitment to space exploration, specifically, to Space Station Freedom, a return to the Moon and, subsequently, a journey to Mars. The material provides a broad, non-technical overview of NASA response to, and support of, President Bush's commitment. In order to hold down the size of the package, a number of charts have been excluded.

Please keep in mind that the Exploration Initiative material is continuously updated as NASA, the National Space Council and others progress in their response to the President's commitment. New charts, as well as material not in this package, is available. Please contact Kristine Johnson (453-9181) or Donna Fabian (453-9177) to see the complete, up-to-date set, and for any assistance.

This "Exploration Initiative Package" is prepared for use by the Office of Exploration and the Office of Aeronautics and Space Technology, but is available to all NASA personnel.

March 1990

Terence T. Finn
Assistant for Policy and External Relations
Office Of Exploration

NASA

NATIONAL SPACE POLICY – GOALS

On November 2, 1989, the President approved a national space policy that updates and reaffirms U.S. goals and activities in space.

- Strengthen the security of the United States
- Obtain scientific, technological, and economic benefits
- Encourage private sector investment
- Promote international cooperative activities
- Maintain freedom of space for all activities
- Expand human presence and activity beyond Earth orbit into the solar system

NASA

VICE PRESIDENT QUAYLE

In May, 1989 the Vice President directed NASA to prepare for a possible major decision on space in a speech by President Bush to be delivered on July 20, 1989

The Vice President called for identification of

- a NASA exploration goal
- significant and visible milestones early in the 21st century
- the resources required (people, facilities, money)

NASA reported to the Vice President that in the final analysis, the nation has but three options for human exploration

- send robots only
- develop a lunar outpost, then go to Mars
- by-pass the Moon and go directly to Mars

NASA

EARTH-MOON-MARS PARAMETERS

The Moon

- 239,000 miles from Earth to Moon
- 1/4 diameter of Earth
- 1/6 Earth's gravity
- Lunar day is 28 Earth days
- Trip time: 3 days one way
- *Launch opportunity every month*
- *Communication time: 2.6 seconds roundtrip*

Mars

- 141.6 million miles from Sun
 - Earth is 93 million miles
- 1/2 diameter of Earth
- 1/3 Earth's gravity
- Martian day is 24 hours 37 minutes
 - Martian year is 1.88 Earth years
- Trip time: 6 months to 1 year one way
- *Launch opportunity every 26 months*
- *Communication time: 10.2-41 minutes roundtrip*

NASA

NASA'S 90-DAY STUDY

In response to the President's speech, the NASA Administrator created a task force, headed by Aaron Cohen, director of the Johnson Space Center, to conduct a study of the main elements of an Exploration Initiative

The study provides reference material in support of the Vice President and the National Space Council, and enables NASA to better understand technical parameters

The study examined

- technical scenarios
- science opportunities
- required technologies
- international considerations
- institutional strengths and needs
- resource estimates

NASA's study consists of analysis, not recommendations. It summarizes extensive trade studies, reflecting several years of study. It is not a definitive program plan

NASA

EXPLORATION APPROACH

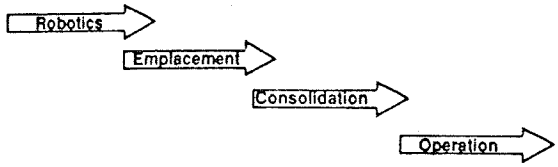
Build upon past and present investments in space
- Apollo, Viking, etc.
- Space Shuttle
- Space Station Freedom

Employ robotic craft along with manned systems

Emphasize science

Build a lunar outpost first
- Research base for science and technology
- Test-bed for humans to Mars

Explore Moon and Mars in phases



NASA

PREREQUISITES FOR HUMAN EXPLORATION

- Exploration technology
- Life sciences research
- Heavy-lift launch and orbiter transfer vehicles
- Robotic missions
- Space Station Freedom

NASA

EXPLORATION HARDWARE NEEDED

Earth-to-orbit launch vehicles

- Space Shuttle
- Existing expendable launch vehicles
- New heavy-lift launch vehicles

Space Station Freedom

- Life sciences research
- Assembly and operations center

Robotic exploration spacecraft

- Design of subsequent human exploration missions
- Technology demonstration

Interplanetary transfer vehicles

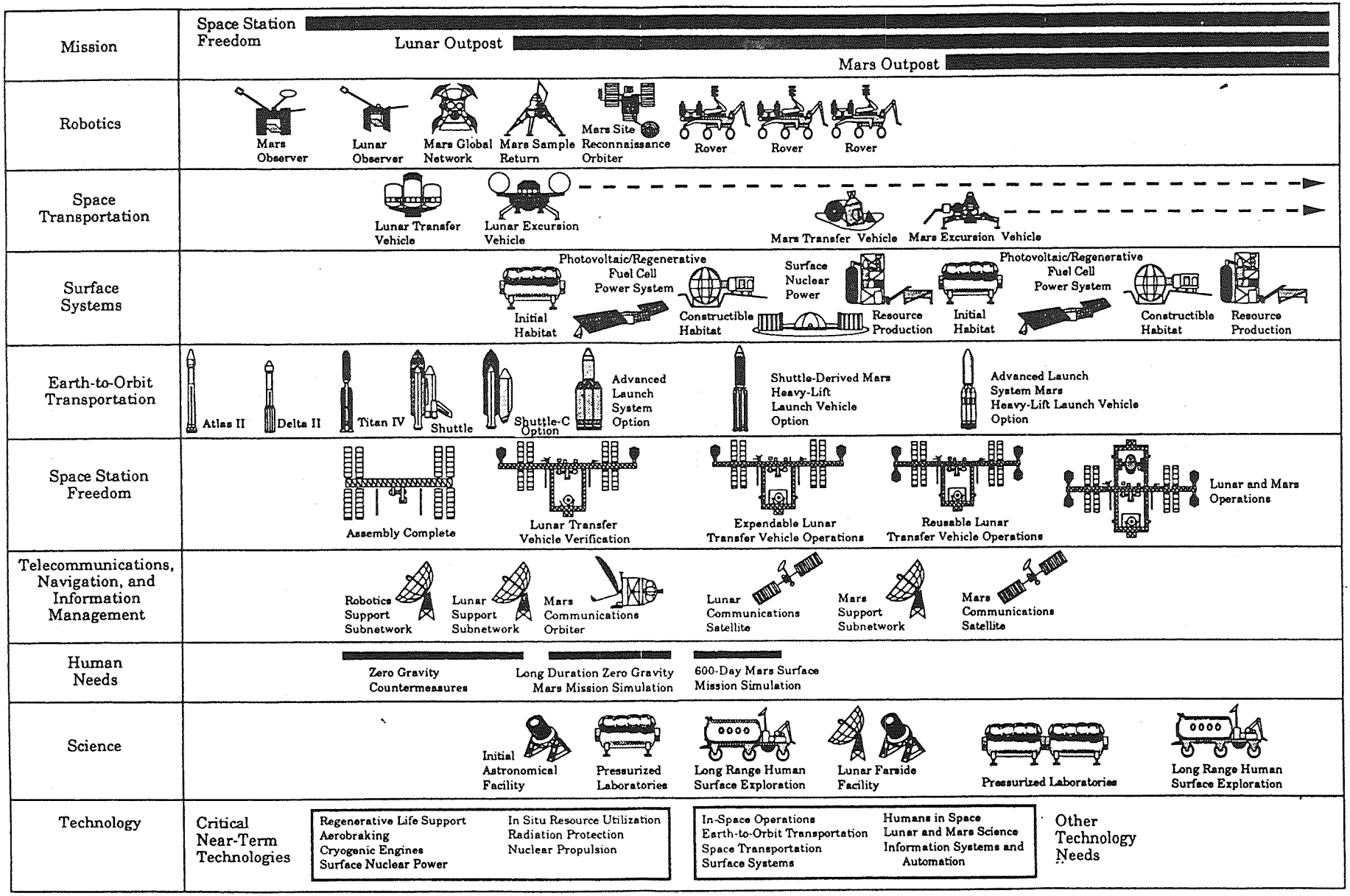
- Transportation between Earth orbit and lunar/Mars orbits

Planetary excursion vehicles

- Transportation between planetary orbit and planetary surface

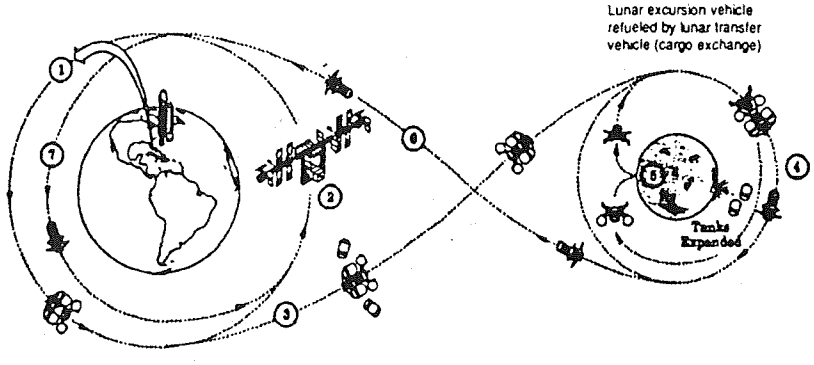
Surface equipment

- Habitats, scientific equipment, rovers, suits, power systems, etc.


Mission
Space Station Freedom
Lunar Outpost
Mars Outpost
Robotics
Mars Observer
Lunar Observer
Mars Global Network
Mars Sample Return
Mars Site Reconnaissance Orbiter
Rover
Rover
Rover
Space Transportation
Lunar Transfer Vehicle
Lunar Excursion Vehicle
Mars Transfer Vehicle
Mars Excursion Vehicle
Surface Systems
Photovoltaic/Regenerative Fuel Cell Power System
Initial Habitat
Constructible Habitat
Surface Nuclear Power
Resource Production
Initial Habitat
Photovoltaic/Regenerative Fuel Cell Power System
Constructible Habitat
Resource Production
Earth-to-Orbit Transportation
Atlas II
Delta II
Titan IV
Shuttle
Shuttle-C Option
Advanced Launch System Option
Shuttle-Derived Mars Heavy-Lift Launch Vehicle Option
Advanced Launch System Mars Heavy-Lift Launch Vehicle Option
Space Station Freedom
Assembly Complete
Lunar Transfer Vehicle Verification
Expendable Lunar Transfer Vehicle Operations
Reusable Lunar Transfer Vehicle Operations
Lunar and Mars Operations
Telecommunications, Navigation, and Information Management
Robotics Support Subnetwork
Lunar Support Subnetwork
Mars Communications Orbiter
Lunar Communications Satellite
Mars Support Subnetwork
Mars Communications Satellite
Human Needs
Zero Gravity Countermeasures
Long Duration Zero Gravity Mars Mission Simulation
600-Day Mars Surface Mission Simulation
Science
Initial Astronomical Facility
Pressurized Laboratories
Long Range Human Surface Exploration
Lunar Farside Facility
Pressurized Laboratories
Long Range Human Surface Exploration
Technology
Critical Near-Term Technologies
Regenerative Life Support
Aerobraking
Cryogenic Engines
Surface Nuclear Power
In Situ Resource Utilization
Radiation Protection
Nuclear Propulsion
In-Space Operations
Earth-to-Orbit Transportation
Space Transportation
Surface Systems
Humans in Space
Lunar and Mars Science
Information Systems and Automation
Other Technology Needs

NASA

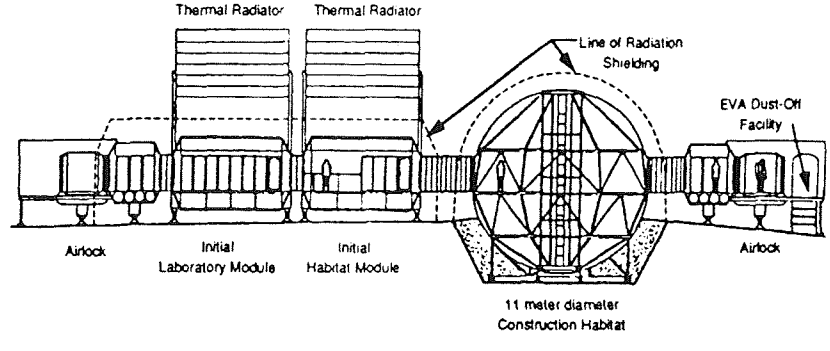
LUNAR MISSION PROFILE



1. Payload Delivered to Space Station Freedom
2. Lunar Transfer Vehicle Mated with Payload at Freedom
3. Trans-Lunar Phase with Lunar Transfer Vehicle
4. Lunar Transfer Vehicle Rendezvous with Lunar Excursion Vehicle from Moon
5. Excursion Vehicle Returns to Moon with Payload
6. Trans-Earth Phase with Transfer Vehicle
7. Transfer Vehicle Aerobrake Maneuver and Return to Freedom

NASA

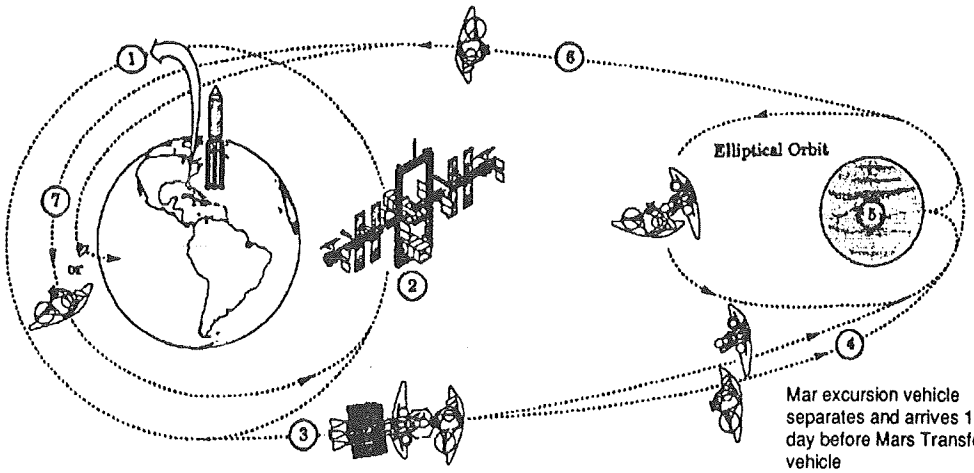
HABITATION FACILITY CROSS SECTION



Space Station-derived modules and inflatable structures

NASA

MARS MISSION PROFILE



1. Payload Delivered to Space Station Freedom
2. Mars Transfer Vehicle Mated with Payload at Freedom
3. Trans-Mars Phase with Lunar Transfer Vehicle
4. Mars Transfer Vehicle Remains in Mars Orbit; Mars Excursion Vehicle Descends to Surface
5. Excursion Vehicle to/from Mars Surface
6. Trans-Earth Phase with Transfer Vehicle
7. Transfer Vehicle Aerobrake Maneuver and Return

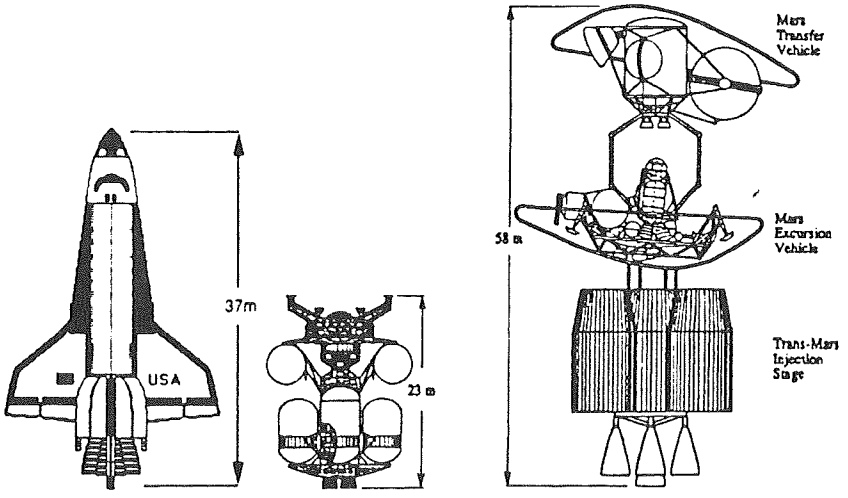
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EARTH-TO-ORBIT TRANSPORTATION

- Lunar outpost and Mars expeditions require large masses in low-Earth orbit 200 → 700 mt/year
- Heavy-lift launch vehicles provide a balance between on-orbit assembly and operations and size of the payloads launched
- Lunar heavy-lift vehicle should provide ~ 70 mt/launch and 3-6 launches per year
- Mars heavy-lift vehicle should provide ~ 140 mt/launch and 3-4 launches per year
- Commercially developed expendable launch vehicles also will be required

NASA

SHUTTLE AND LUNAR/MARS TRANSFER VEHICLES



Space Shuttle
Mass = 92 metric tons
(Payload = 22 metric tons)

Lunar Transportation System
Mass = 200 metric tons

Mars Transportation System
Mass = 800 metric tons

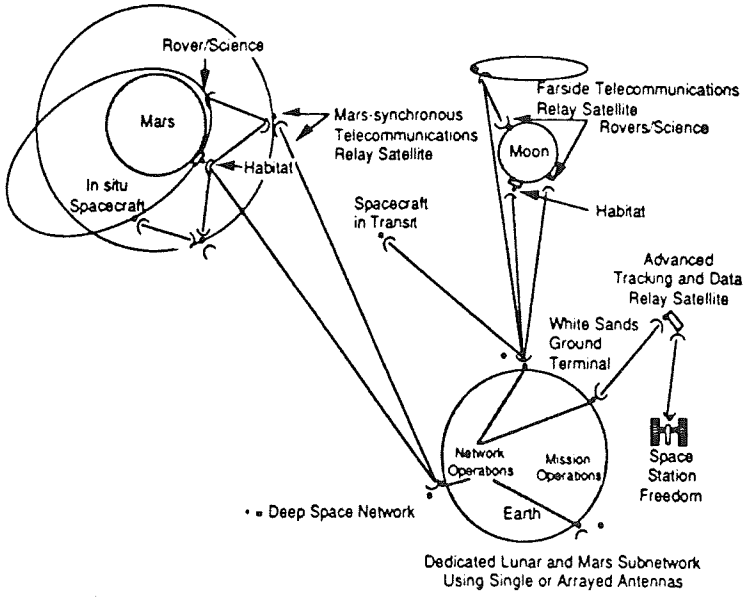
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CHARACTERISTICS OF REFERENCE APPROACHES

	Reference Approaches				
	A	B	C	D	E
	Earliest Moon			No Space Station Disruption	
	Earliest Lunar Outpost	Earliest Mars	Later Mars	Permanently Manned Lunar Outpost	Man-Tended Lunar Outpost
Lunar Emplacement	1999-2004	1999-2004	1999-2004	2002-2007	2002-2007
Lunar Consolidation	2004-2009	2004-2007	2004-2008	2007-2012	2008-2013
Lunar Operation	2010→	2005→	2005→	2013→	2014→
Humans on the Moon	2001	2001	2001	2004	2004
Permanent Habitation	2002	2002	2002	2005	→
Constructible Habitat	2005	2006	2007	2008	2011
Eight Crew	2006	2007	2007	2009	—
Lunar Oxygen Use	2010	2005	2005	2013	—
Lunar Farside Sortie	2012	2008	2008	2015	2022
Lunar Steady State Mode	2012	2008	2012	2015	→
Mars Emplacement	2015-2019	2010-2015	2015-2019	2017-2022	2024→
Mars Consolidation	2020-2022	2015-2018	2020-2022	2022	→
Mars Operation	2022→	2018→	2022→	→	→
Humans on Mars	2016	2011	2016	2018	2016
Extended Mars Stay	2018	2014	2018	2023	2027

NASA

TELECOMMUNICATIONS ARCHITECTURE FOR THE HUMAN EXPLORATION INITIATIVE



NASA

CONCLUSIONS OF THE 90-DAY STUDY

- Major investments in challenging technologies are required
- Scientific opportunities are considerable
- Robotic spacecraft will be needed
- Current launch capabilities are inadequate
- Space Station Freedom is essential
- Program alternatives do exist
- Opportunities for international cooperation exist
- A long-range commitment and significant resources will be required

NASA

SPACE STATION FREEDOM

A permanently manned, international research laboratory and, later, a staging base for the Moon and Mars

Need for:

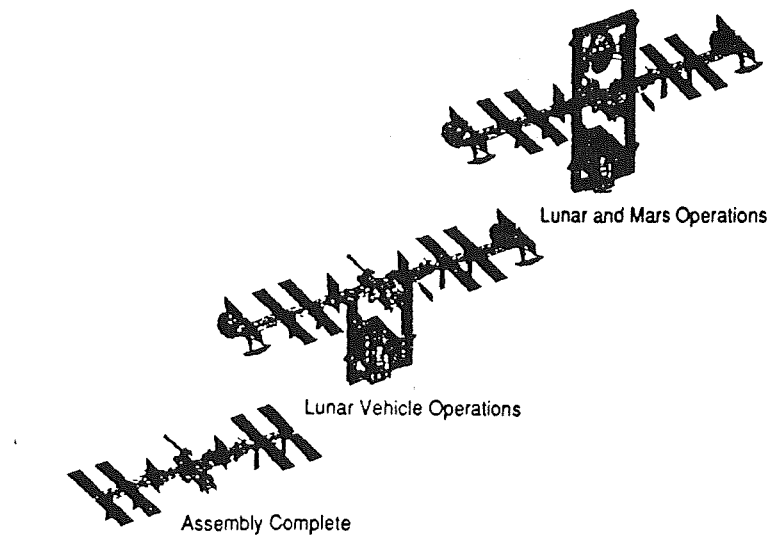
- Life sciences research and microgravity countermeasures
- Technology development and validation
- Development of operational procedures
- Assembly, test, launch, recovery, turnaround of space vehicles

Current design can evolve to the more capable configuration essential for a return to the Moon and human exploration of Mars

President Bush called Space Station Freedom: "our critical next step in all our space endeavors"

NASA

SPACE STATION FREEDOM
EVOLUTION FOR HUMAN EXPLORATION



NASA

STATION AND EVOLUTION

In 1984 President Reagan called for a station that was
- a research facility
- permanently manned
- international in character

Freedom's assembly and operations have made it a transportation node from the very beginning

Freedom's multi-disciplinary research capabilities are a balance between microgravity environment and the need for human presence

Freedom can evolve to support the Exploration Initiative
- additional required resources to be phased in
- international agreements will be honored. Exploration enhancements to come out of U.S. allocation
- hooks and scars on *Freedom* must be protected

Earth-to-Orbit logistic requirements are drivers on transportation node

Current configuration is the correct design for both near-term and later requirements

NASA

HEAVY LIFT LAUNCH VEHICLE

Exploration Initiative requires enhancement of current launch vehicle capabilities

Earth-to-orbit lift capabilities are estimated to be:

- Moon: 60 - 70 metric tons
- Mars: 140 metric tons

New launch vehicle development candidates include:

- Shuttle-C
- Advanced Launch System

There are no major technical impediments to building the heavy-lift launch vehicles we need

Expendable vehicles to play key role in Exploration Initiative

We ought to be initiating development

NASA

SPACE RESEARCH AND TECHNOLOGY

Program Goals

- Increase safety and reliability
- Reduce development
- Enhance mission performance
- Enable new missions

Program Elements

- Research and Technology Base
- Civil Space Technology Initiative (CSTI)
- Exploration Technology Program
- In-Space Technology Experiment Program (INSTEP)

NASA

EXPLORATION TECHNOLOGY MISSION APPLICATIONS SUMMARY

Technology Thrust	Technology Program Area	Moon Robotic	Moon Human	Mars Robotic	Mars Human	Other Solar System Exploration
Space Transportation	Aerobraking		●	●	●	◗
	Space-based Engines		●	○	●	○
	Autonomous Vehicles Maneuvering		◗	◗	◗	◗
In-Space Operations	Cryogenic Fluid Systems		◗		◗	
	In-Space Assembly and Construction		◗		◗	○
	Vehicle Servicing and Processing		◗		◗	○
Surface Operations	Space Nuclear Power	◗	●	◗	●	◗
	In Situ Resource Utilization		●	◗	●	
	Planetary Rover	◗	◗	◗	◗	◗
	Surface Solar Power	○	◗	○	◗	◗
	Surface Habitats and Construction		◗		◗	
Human Support	Regenerative Life Support		●		●	
	Radiation Protection		●		●	
	Extravehicular Activity/Suit		◗		◗	
	Space Human Factors		○		○	
Lunar and Mars Science	Sample Acq. Analysis, and Preserv.	◗	○	◗	○	◗
	Probes and Penetrators	○		◗	○	◗
Information Systems and Automation	High-Rate Communications	○	○	◗	◗	◗
	Exploration Automation and Robotics	◗	◗	◗	◗	◗
	Planetary Photonics	○	○	◗	○	◗
Nuclear Propulsion	Nuclear Thermal Propulsion				●	◗
	Nuclear Electric Propulsion				●	◗
Innovative Technologies and Systems Analysis		○	○	○	○	○

LEGEND ○ High-Leverage Technology ◗ Enabling For Some Systems ● Critical Technology

NASA

SOME CONCRETE EXAMPLES

EXPLORATION	→ Aeroassist flight experiment
TRANSPORTATION	→ Structural analyses for solid rocket motor redesign
SPACE STATION	→ Erectable truss structure
SPACE SCIENCE	→ Silicon CCD area arrays for Space Telescope
BREAK THROUGH	→ Photonics (optical processors)

These are but a few examples of successful products developed by NASA's space research and technology program

NASA

ADDITIONAL EXAMPLES: R&T PRODUCTS FOR SPACE SCIENCE

- Spacecraft ground operations automation — *Voyager*
- Deviser planner — *Voyager and Galileo*
- Advanced TWT amps. and low noise receivers — *CTS, ACTS, Mariner Mars Observer*
- Massively parallel processor — *Climate modeling*
- Millimeter accuracy laser ranging system — *LAGEOS*
- Spacecraft charging model — *GSFC, JPL, Industry*
- High power/voltage transistors — *Industry*
- SAR technology — *SeaSat, SIR (A, B, and C)*
- Heat shield design and analysis — *Galileo probe*
- Silicon CCD area arrays — *Hubble Space Telescope, Galileo*
- Fiber optics rotational sensor — *CRAF/Cassini*
- X-band uplink-down converter — *Galileo*
- Advanced digital SAR processor — *Magellan*
- IR sensors — *SIRTF instruments*
- CO_2 laser — *EOS, LAWS*

NASA's space research and technology program is also developing products in the fields of space transportation, space station, exploration, as well as "breakthrough" fields where payoffs would be extremely high

NASA

TECHNOLOGY REPORT TO CONGRESS

Concerns: NASA's space technology programs not sufficiently focused to meet the needs of long-term space exploration as outlined in the President's speech in July 1989.

Requirement: *Provide a report by February 1, 1990** on specific technologies needed to meet the development and operational requirements of the President's space exploration initiative.

- Prioritize technologies both technically and financially
- Include five-year funding profile

Source: Senate Appropriations Committee Report accompanying H.R. 2916, the Departments of Veterans Affairs and Housing and Urban Development, and Independent Agencies Appropriation Bill, 1990, page 113.

* Targeting for March 1

NASA

EXPLORATION LIFE SCIENCES

- Radiation Protection
- Reduced Gravity Countermeasures
- Life Support in Habitats and Space Vehicles
- Extravehicular Activity
- Medical Care
- Behavior and Performance

Earth ➡ Freedom ➡ Lunar Outpost ➡ Mars

NASA

RADIATION PROTECTION

Earth's magnetic field protects us from radiation emitted by solar flares, and shields us from galactic cosmic rays

Radiation beyond Earth orbit is cause for concern

Radiation strategy for the Exploration Initiative includes

- determination of career dose limits and crew selection criteria
- development of countermeasures
- development of shielding strategy for both vehicles and habitats
- development of early warning systems and "storm shelters" for protection from solar flare radiation

NASA will develop guidelines with the National Council on Radiation Protection and Measurements

- NASA will adhere to the radiation principle of as low as reasonably achievable (ALARA)

NASA

ARTIFICIAL GRAVITY?

Microgravity exposure causes major physiological change

- Bone mineral loss
- Muscle atrophy
- Cardiac deconditioning

Current countermeasures (exercise) may be insufficient for the lengthy voyage to Mars

Strategy to test and evaluate necessary zero-g countermeasures will utilize

- Soviet long duration experience
- Space Shuttle extended duration orbiter
- Space Station Freedom and eventually
- The lunar outpost itself

Current approach: plan a zero-g Mars transfer vehicle, but begin low level definition of an artificial gravity system just in case

Humans must be certified for journey to Mars

NASA

SCIENCE: SIGNIFICANT OPPORTUNITIES

Excellent science to be done on both Moon and Mars

- Robotic science
- Human interactive science

Fundamental scientific themes

- Origin and history of Earth and Moon
- The origin of life/life on Mars
- Global climate change
- Search for other solar systems
- Fate of the Universe

Research opportunities cover many disciplines

- Solar Physics
- Geology
- Biology
- Astrophysics
- Chemistry
- Space Physics

NASA

ROBOTIC SPACECRAFT

Key tasks

- Determine suitable/desirable landing and outpost sites
- Provide design data for human mission elements
- Conduct science investigations
- Develop basis of science investigations for human explorers

Select from high payoff candidate missions

For the Moon, emphasis on selecting landing/outpost site

- Lunar Observer

For Mars, emphasis on science and human mission success

- Mars Observer
- Global Network Mission
- Sample Return/Local Rover
- Site Reconnaissance Orbiter
- Mars Rovers

Robotic missions are integral to human exploration initiative

NASA

SCIENCE ON THE MOON

Lunar origin/evolution
- Impact origin theory ⇨ common origin with Earth
- Larger role for planetary scale collisions?

History of the Sun (preserved in lunar soil)
- Solar wind trapped in regolith
- Buried regolith provides time resolution

Extinctions caused by impacts
- Evidence in lunar cratering record?

Unparalleled resolution, sensitivity for astronomy/astrophysics
- Large apertures
- Interferometric arrays
- Cosmic Ray Observatory

Life science
- Basic research: radiation environment, low gravity effects ...
- Supporting Mars exploration

NASA

SCIENCE ON MARS

Planet most like Earth
- Has an atmosphere, evidence of warmer past
- Mars has intrigued humans for generations

Search for life on Mars
- Life may have existed long ago
- It may still exist in protected underground environments
- Answers will provide clues about evolution of life

Global climate change on Mars
- Examine chronology, characteristics of changes
- Understand role of geologic processes (e.g., volcanism, weathering)
- May enhance our understanding of changes on Earth

Human and robotic exploration
- Both important for complex field studies

Human presence key to advancing understanding

NASA

ROBOTIC MISSIONS TO MARS

Purpose

- Secure a better understanding of the planet
- Provide data to assist in designing manned systems
- Support selection and certification of outpost sites
- Return sample for scientific analysis
- Demonstrate readiness to proceed with human missions

Missions

- 1992 Mars Observer
 - Establish global data base
- Mars Global Network Mission
 - Employ landers to provide high-resolution surface data
- Mars Sample Return Mission (MSRM)
 - Return samples for analysis
- Mars Site Reconnaissance Orbiter
 - Provide details to characterize landing sites
- Mars Rover Mission
 - Certify sites and explore the planet's surface

NASA

INTERNATIONAL COOPERATION

Precedents are mixed

- Apollo/Viking: U.S. only
- Space Shuttle: primarily U.S.
- Space Station Freedom: international partnership
- Hubble Space Telescope: international participation

Advantages are significant

- Access to first-rate technical capabilities
- Reduction in costs
- Stronger ties with other nations
- Foreign resources tied to U.S. initiative

Disadvantages not to be discounted

- Dilution of control
- Management complexity
- Reduced U.S. leadership
- Vulnerable to political climate

Significant opportunities exist

NASA

INTERNATIONAL COOPERATION

Japan

- Limited experience
- Ambitious aspirations
- Growing capabilities: H-II vehicle and Space Station module
- Interested in lunar resource utilization

Europe

- Technically expert
- Seeking autonomous capabilities in manned space flight
- Partner in Space Station, and designing Hermes space plane
- Would seek equality in any future initiative

U.S.S.R.

- Returned lunar samples robotically in 1970s
- Active planetary program, had focused on Venus
- Interest in Mars, but limited success
- Proposed a manned Mars project with the U.S.

Canada

- Built Canadarm for Shuttle
- Building Mobile Servicing System for Space Station Freedom
- Significant robotic capabilities
- Would probably welcome a role in this area

Other Nations

- China, India and Brazil have small space programs
- Desire to participate?
- Role for nations with small or no space experience?

NASA

EXPLORATION OUTREACH ACTIVITY

- In a September 1989 letter to the Vice President, the NASA Administrator said the agency would explore a complete range of options including technologies and mission architectures upon completion of the 90-Day Report
- In a December 1989 letter to Admiral Truly, the Vice President requested NASA take the lead in a nationwide search for new ideas and innovative technologies "to ensure all reasonable space exploration alternatives have been evaluated."
- Responding in late January 1990, the Administrator Truly wrote that NASA would do so, employing "an array of formal and informal mechanisms to reach the widest segment possible of the American scientific and technological communities"
- Likely mechanisms will include NASA Research Announcements (NRA), site visits and reviews with national laboratories and other agencies, aerospace industry analyses, AIAA assessment and conference, and direct solicitation of professional societies and individuals
- NASA will incorporate a review mechanism, with participation from outside the agency, to select promising ideas and technologies for funding in FY 1991
- Reviews by, and discussions with, the National Research Council, the NASA Advisory Council and the National Space Council will be part and parcel of the outreach activity

NASA

CURRENT ACTIVITIES

- Working with the National Space Council staff to structure a nation-wide outreach program to search for technical innovations and new ideas
- Merging Office of Aeronautics and Space Technology and Office of Exploration
- Continuing our preliminary science planning in conjunction with the Office of Space Science and Applications
- Developing implementation plans for exploration technology initiatives
- Planning exploration mission studies
- Working with National Space Council staff in support of Council recommendations regarding international affairs
- Supporting National Research Council (NRC) and Aerospace Industries Association (AIA) reviews of the Exploration Initiative

NASA

NATIONAL SPACE COUNCIL

Mandated in FY 1989 NASA Authorization Act and established pursuant to an Executive Order signed April 20, 1989

Purpose: "to provide a coordinated process for developing a national space policy and strategy for monitoring its implementation"

Members: Vice President - Chairman

- Secretary of State
- Secretary of the Treasury
- Secretary of Defense
- Secretary of Commerce
- Secretary of Transportation
- Director of the Office of Management and Budget
- Chief of Staff to the President
- Assistant to the President for National Security Affairs
- Assistant to the President for Science and Technology
- Director of Central Intelligence
- Administrator of NASA

NASA is currently supporting the Council's efforts to develop decision packages for the President on a human exploration strategy.

NASA

TWO INDEPENDENT REVIEWS

The National Space Council has requested two independent reviews of the Exploration Initiative:

AEROSPACE INDUSTRIES ASSOCIATION (AIA)

- chaired by Jim Harrington of Kamen Aerospace Corp.
- looking at strategy and the process for implementing the Exploration Initiative
- to recommend a management methodology
- targeting late March, 1990 for completing report

NATIONAL RESEARCH COUNCIL (NRC)

- chaired by Guy Stever of the National Academy of Sciences
- looking at the scope and content of NASA's 90-Day Report
- to address technical assumptions, alternative technologies, and schedule/cost considerations
- targeting late February, 1990 for completing report

NASA supportive of both AIA and NRC Studies

NASA

SOME CONCLUDING THOUGHTS

NASA will support National Space Council activities and welcomes independent external reviews of the Exploration Initiative

Outreach for new ideas and new technologies will be broad in scope

Near-term NASA focus will be on

- technology strategies
- mission architecture
- planning for science

The Space Station Freedom program must receive full support

This is a "long-term, continuing commitment" and all of us must be prepared for a lengthy period of planning and policy development

NASA

THE PRESIDENT STATES THE GOAL

"Our goal: To place Americans on Mars—and to do it within the working lifetimes of scientists and engineers who will be recruited for the effort today. And just as Jefferson sent Lewis and Clark to open the continent, our commitment to the Moon/Mars initiative will open the Universe. It's the opportunity of a lifetime—and offers a lifetime of opportunity."

President George Bush
Remarks at the University of Tennessee
February 2, 1990

PRESENTATION 1.1.2

NATIONAL SPACE TRANSPORTATION STRATEGY

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N91-28195

NASA'S ADVANCED SPACE TRANSPORTATION SYSTEM LAUNCH VEHICLES

Darrell R. Branscome
Director, Advanced Program Development Division,
National Aeronautics and Space Administration

NASA'S ADVANCED SPACE TRANSPORTATION SYSTEM LAUNCH VEHICLES

Darrell R. Branscome
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Office of Space Flight
National Aeronautics and Space Administration
Washington, DC 20546

ABSTRACT

On July 20, 1989, the 20th anniversary of the first Apollo-Lunar landing, President Bush outlined a long term national program for the Human Exploration of the Moon and Mars. Building upon the capabilities provided by Space Station Freedom, the President envisioned returning to the Moon and establishing a permanent manned station, to be followed by manned mission to Mars early in the next century. These are bold, new goals for the U.S. Space Program. They are, however, built upon a solid and pragmmatic base of planning. These demanding but realistic mission objectives, reflect the highest technical and enginering capabilities residing within the government and industrial capabilities of the industry.

This paper will provide some insight into the advanced transportation planning and systems that will evolve to support these long-term mission requirements. The general requirements include: launch and lift capacity to low earth orbit (LEO); space-based transfer systems for orbital operations between LEO and geosynchronous equatorial orbit (GEO), the Moon, and Mars; and transfer vehicle systems for long duration deep-space probes. These mission requirements are incorporated in the NASA Civil Needs Data Base. To accomplish these mission goals, adequate lift capacity to LEO must be available: to support science and application missions, to provide for construction of the Space Station Freedom and to support resupply of personnel and supplies for its operations. Growth in lift capacity must be time-phased to support an expanding mission model that includes Freedom Station, the "Mission To Planet Earth", and an expanded robotic planetary program. Near term launch vehicle system improvements will capitalize on the existing hardware and infrastructure of the Shuttle.

The near term increase in cargo lift capacity associated with development of the Shuttle-C vehicle will be addressed. The joint DOD/NASA

Advanced Launch System studies are focused on a longer term new cargo capability that will significantly reduce costs of placing payloads in space.

Longer term transportation studies include the Next Manned Transportation System, and Space Transfer Vehicles. The Next Manned Transportation System studies are focused on concepts to extend, complement, or replace the Shuttle after the turn of the century. The next manned transportation system assessment is focussed on three distinctly different paths: Shuttle Evolution, a new Personnel Launch System, or an Advanced Manned Launch System. Space Transfer Vehicle studies to satisfy robotic and human exploration missions also have been initiated.

Activation of Space Station Freedom in the mid-90's connotes continuous human habitation with increasing crew complements and activities over time. If an accident were to occur, or if a major medical emergency were to arise, there must be an assured crew return capability. NASA has initiated a program to address and evaluate the vehicle options and systems implications associated with providing this capability. Several contracted Assured Crew Return Vehicle concepts are under study and will be described.

All of these transportation vehicle activities are inter-related, and time-phased to provide a comprehensive planning base for decisions related to future elements of national space transportation capabilities. These programs provide broad options in terms of technology, cost, and development risk, and in terms of fleet size, lift capacity, and mission operational flexibility. When combined with companion studies on missions and experiments, a complete set of program options will be available for defining the course of the United States civil space program.

Introduction

President Bush, during the 20th Anniversary of the First Manned Landing on the Moon cermonies, recognizing that the Space Shuttle has returned to flight and that the development of the international Space Station Freedom is now underway, established a long term national goal for the United States to lead a program directed to the Human Exploration of the Moon and Mars. These missions are the realization of mission planners "dreams" from the earliest days of the U.S. Space Program. While these mission are extremely challenging and will demand the ultimate in engineering and science capabilities and skills, they are achievable and are, in fact, the culmination of planning and study activities that have been underway for over six years in anticipation of these decisions.

During this period the United States has substantially altered the proposed content of our future National Space Program. These changes began in 1984 with President Reagan's "State of the Union" announcement of the decision to establish a permanent manned presence in space using an international space station[1]. Also during 1984, at the direction of the United States Congress, the National Commission on Space, was formed to review the U. S. space program, to recommend long range goals, and to define a roadmap for the next fifty years. In their report, published in May 1986, *Pioneering The Space Frontier*, the Commission recommended an orderly, step-by-step program, based on a broad expansion and development of low cost institutions and operating systems, which would ultimately lead to the exploration of the solar system and habitation of the Moon and Mars[2].

The Commission's plan for low cost access to the inner solar system has been replicated as **Figure 1**. The first section, *Highway to Space*, outlines the transportation requirements to Earth orbit and for orbital operations. Cargo and passenger transport vehicles are identified as well as transfer vehicles having the ability to base at Space Station Freedom. The second section, *Bridge Between Worlds*, identifies expansion of operations beyond Earth orbit. Large transfer vehicles are envisioned, operating between Earth's orbit and the lunar and Mars orbits, followed by surface operations and extended surface habitation.

Subsequent to the Commission's Report, the NASA Administrator formed a study team, chaired by Astronaut Sally Ride, to define an implementation plan for the achievement of a national space policy directed toward an expanded human presence in space. The Ride report to the NASA Administrator, *Leadership and America's Future in Space*[3], recommended four major mission elements::

- Mission To Planet Earth
- Exploration of the Solar System
- Outpost on the Moon
- Humans to Mars

The report provided a roadmap for the President Bush's Human Exploration Initiative and is the framework for detailed long range NASA planning activities.

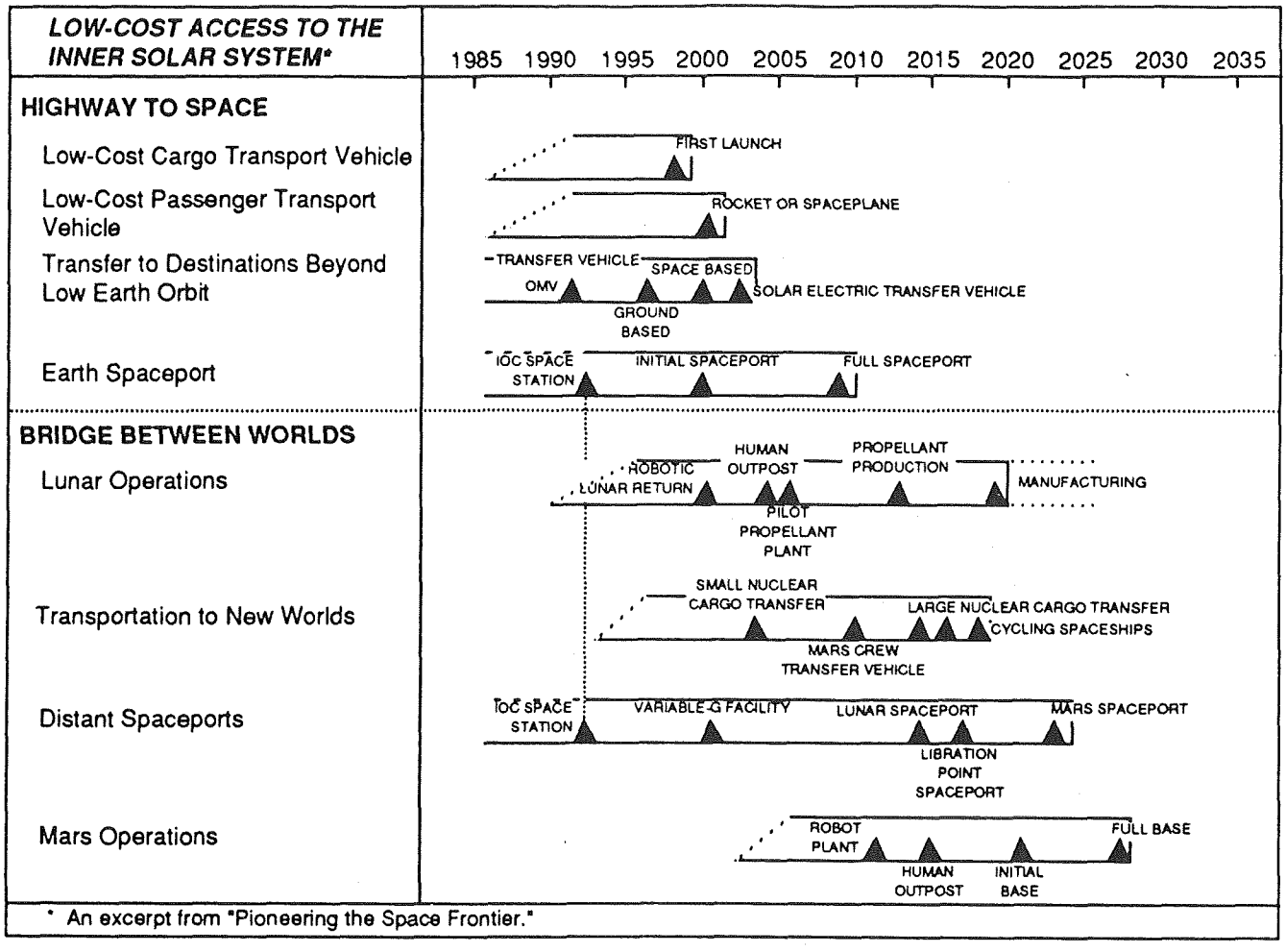


Figure 1. Low-Cost Access to the Inner Solar System.

Background

Both civilian and military space program plans were affected by national space policy decisions that occurred during the 1980's. To map out an orderly and balanced plan for the United States to follow, the Joint NASA/DOD National Space Transportation Support Study (NSTSS) was initiated. This study program, often referenced as the Space Transportation Architecture Study (STAS), established overall space transportation needs and defined timeframes when these capabilities would be required. As outlined in **Table 1**, the STAS Study Team recommended that five major capabilities be phased in by 2005. The requirement for a cargo return vehicle has been satisfied by modification of the Shutle to provide increased downweight and landing capability. The recent LDEF recovery of approximately 22,000 lbs.was a record for landed Shuttle weights. A requirement to provide for an Assured Crew Return Capability (ACRC) from Space Station Freedom was subsequently added by NASA advisory bodies.

Mission Requirements Definition

For the remainder of the century, the United States' civil programs will rely in large part on the Shuttle to transport all personnel and most large payloads to orbit. Major mission requirements, as summarized in **Table 2**, illustrate the significant increases in launch demand over time. Near-term launch requirements are dominated by the delivery of science and solar system exploration spacecraft, Spacelab, and a variety of DOD payloads. In the period from the mid-1990's through 2000, the assembly, activation, and crew exchange for Space Station Freedom and launch of the Earth Orbiting System-Polar Orbiting Platforms significantly increase launch requirements. Beyond the turn of the century, sustaining crew rotation and logistic support of Space Station Freedom operations, science observatories, robotic planetary explorers, and human exploration initiatives will require additional transportation capabilities.

In order to match the wide variety of payload manifesting requirements to projected launch capacity and schedules, NASA has developed the Civil Needs Data Base (CNDB). The CNDB provides insight into the total annual mass to be delivered and the numbers of payloads that will require delivery to specific orbital locations. The CNDB is revised annually to project all future civil mission requirements[4]. Two models are developed within the CNDB as illustrated in **Figure 2**; a base model, and an expanded model reflecting increasing levels of program activity.

Table 1

SUMMARY OF NATIONAL SPACE TRANSPORTATION SUPPORT STUDY RECOMMENDATIONS

SUMMARY OF NSTSS STUDY TEAM RECOMMENDATIONS

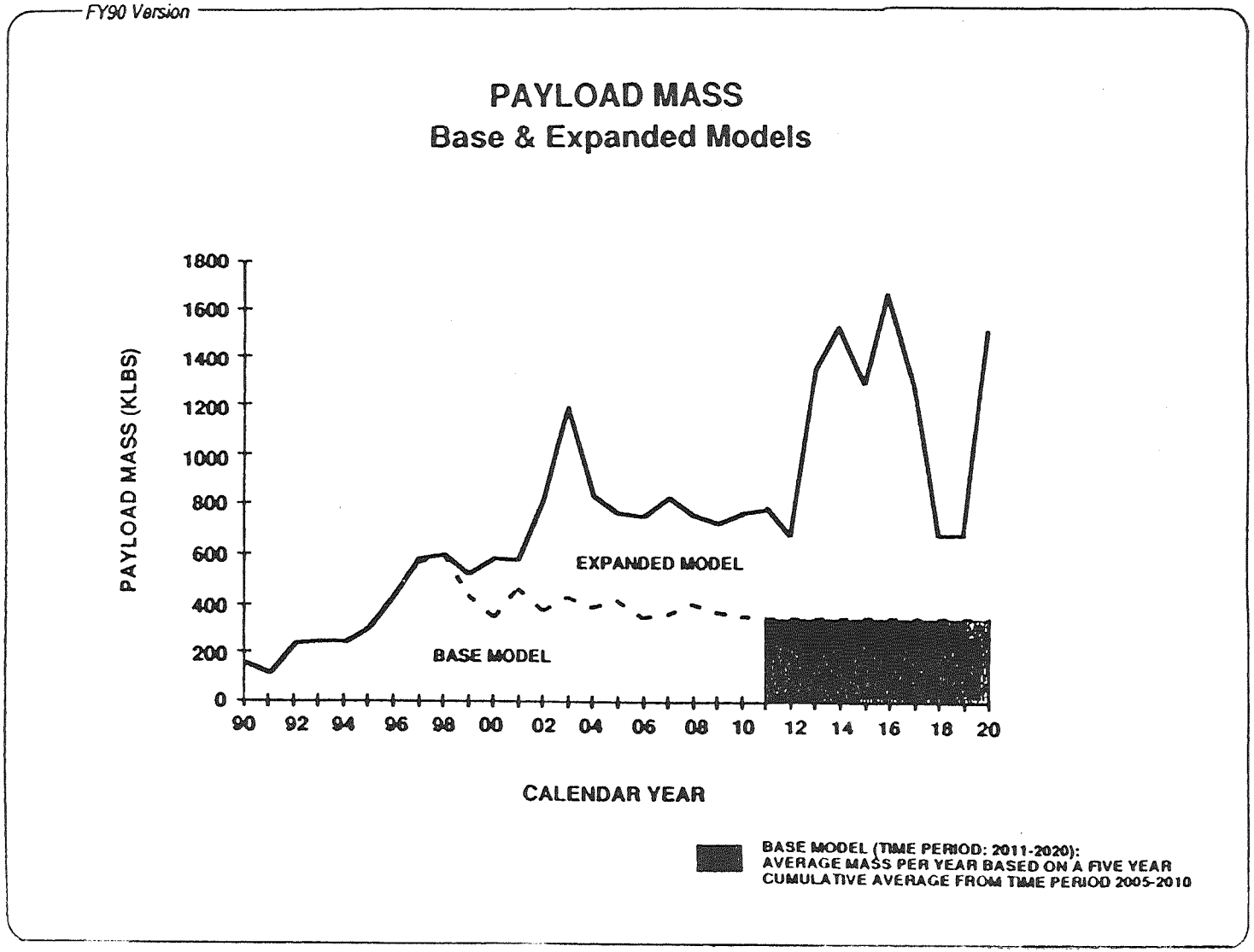
	TIME FRAME
• Cargo Vehicle / Heavy Lift	
- DERIVATIVE VEHICLE	Mid 1990's
- NEW VEHICLE	~ 2000
• Next Manned Space Transportation System	2005
• New High Energy Upper Stage	~ 2000
• Cargo Return Vehicle	Mid-1990's

Table 2

MISSION REQUIREMENTS (SUMMARY)

	Near-Term
	• Science Observatories - HUBBLE SPACE TELESCOPE - GAMMA RAY OBSERVATORY - ADVANCED X RAY ASTRONOMY FACILITY
Current Launch Requirements	• Solar System Exploration - GALILEO - ULYSSES - MARS OBSERVERS
	• Spacelab
	• DoD Missions
	Mid-1990's - 2000
	• Space Station Freedom Assembly
Growing Launch Requirements	• Space Station Freedom Logistics - RESUPPLY - CREW EXCHANGE
	• Polar Platforms / Mission To Planet Earth
	2000 And Beyond
	• Space Station Freedom Growth / Logistics
Expanded Launch Requirements	• Science Observations
	• Robotic Planetary Exploration
	• Human Exploration

Figure 2


FY90 Version
PAYLOAD MASS
Base & Expanded Models
PAYLOAD MASS (KLBS)
1800
1600
1400
1200
1000
800
600
400
200
0
EXPANDED MODEL
BASE MODEL
90 92 94 96 98 00 02 04 06 08 10 12 14 16 18 20
CALENDAR YEAR
BASE MODEL (TIME PERIOD: 2011-2020):
AVERAGE MASS PER YEAR BASED ON A FIVE YEAR
CUMULATIVE AVERAGE FROM TIME PERIOD 2005-2010

The base model is developed by summing the specific mission needs of each current NASA Program Office. These include misions contained in the NASA Mixed Fleet Mainifest, all Space Station Freedom assembly/ construction, crew rotation and logistic support, Mission to Planet Earth and the deep space program launches.

The expanded model includes projected requirements and includes launch mass additions for conversion of the Space Station Freedom from a micro-gravity facility to a transportation node, the deep space science payloads with high-energy stages required for the unmanned precursor missions to the planets, and the Lunar and Mars human exploration mission now being conceptualized by the Office of Exploration in NASA. Significant total mass increases of the expanded model over the base model after the turn of the century are apparent.

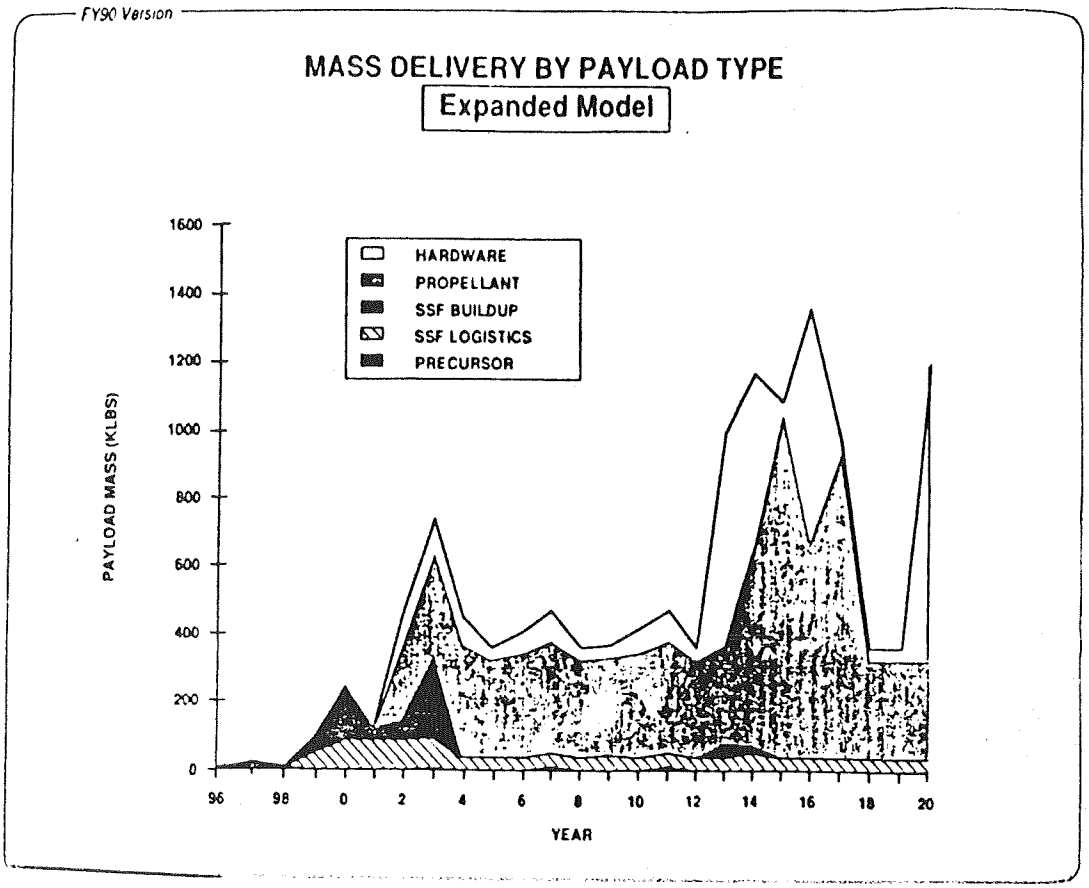
A detailed review of the expanded model also clearly illustrates projected differences in future launch vehicle requirements from capabilities currently available in the Shuttle and ELV's. As shown in **Figure 3**, hardware and propellant launch requirements for the manned Lunar and Mars missions after the turn of the century literally overwhelm all of the other requirements for Space Station buildup, Space Station logistics, and the planetary precursor missions, immediately . These are the data necessary for planning and sizing the future US launch vehicle fleet

Cargo Vehicle Definition Studies

The United States has a clear and evolving need for increased lift capacity to deliver both large masses and large volumes to LEO. Mission requirements in the CNDB indicate that a large, unmanned, cargo launch vehicle is necessary and could satisfy a "niche" in the total launch vehicle inventory later in this decade and into the next Century. Development of unmanned cargo vehicles with payloads in the range of 100K-300K pounds to LEO, using either existing assets or new technology, would be extremely cost effective. Increases in the cargo payload per launch could be applied to reduce the total number of launches required and to reduce and simplify the orbital assembly operations mandated by small, multiple units of structure. Larger structural units, tanks, and fuel supplies for energetic planetary missions could be delivered in fewer flights. Two large cargo vehicle concepts are being explored. Either one can provide the United States with a wide range of payload mass and volume options.

The unmanned Shuttle-C launch vehicle concept, which makes use of existing Shuttle elements and infrastructure, could be available in the mid-1990's. In Shuttle-C, shown in **Figure 4**, the Orbiter would be replaced

Figure 3


FY90 Version
MASS DELIVERY BY PAYLOAD TYPE
Expanded Model
HARDWARE
PROPELLANT
SSF BUILDUP
SSF LOGISTICS
PRECURSOR
PAYLOAD MASS (KLBS)
1600
1400
1200
1000
800
600
400
200
0
96
98
0
2
4
6
8
10
12
14
16
18
20
YEAR

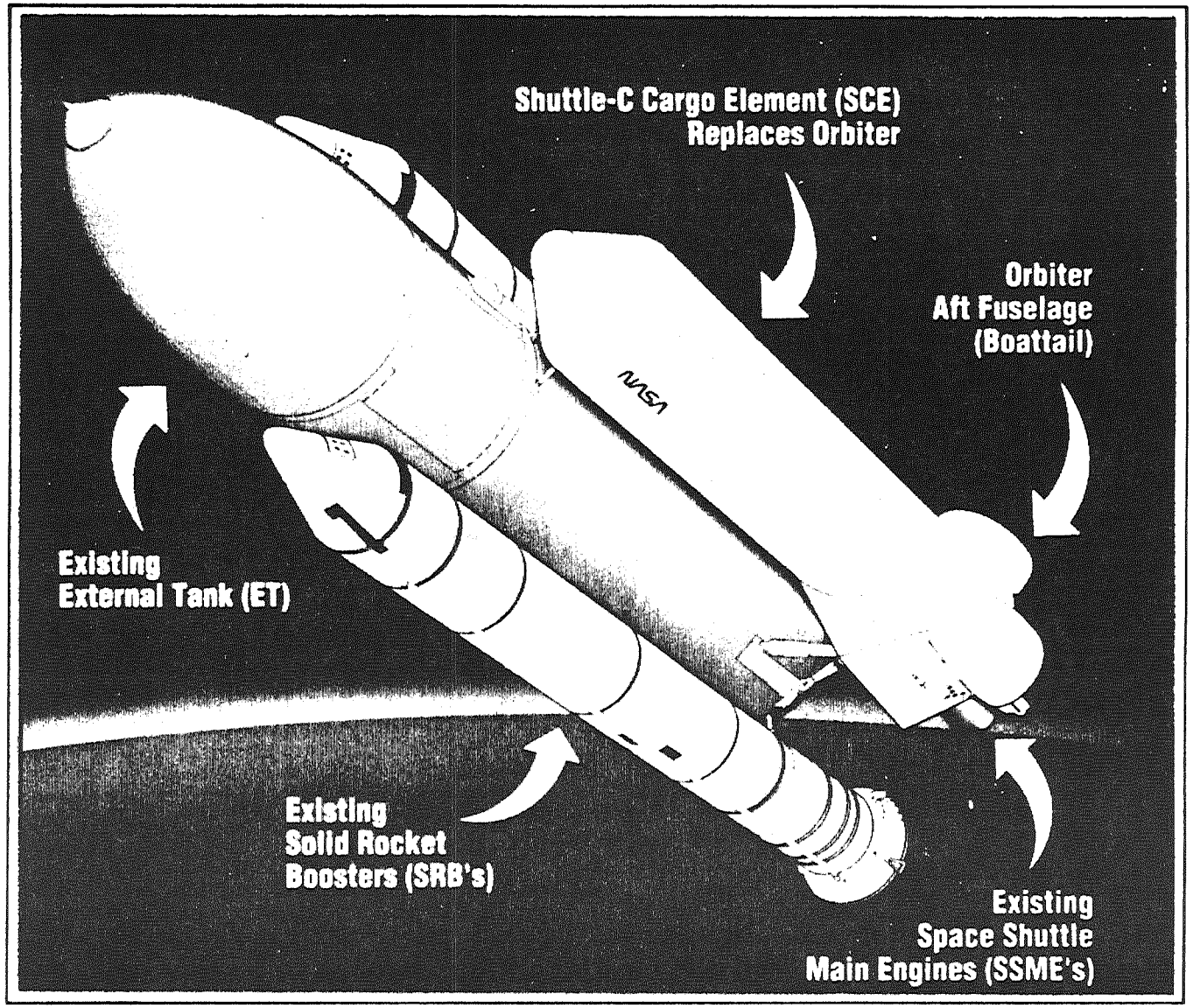


Figure 4 Shuttle-C Elements.

with a large cargo carrier element mounted in the same location. This new cargo element design is illustrated in the full sized Engineering Model at the Marshall Space Flight Center (Figure 5) and has an aerodynamic nose fairing on the forebody and a modified and simplified orbiter afterbody with three Space Shutle Main Engines. The payload is mounted internally under full length split doors which open for deployment at LEO injection altitude. On completion of the operation, the cargo element structure including the engines reenters the atmosphere. The Shuttle-C program schedule is shown in Figure 6. Shuttle-C offers the potential for lifting 100K-150K pounds to orbit. Shuttle-C, operating concurrently with Shuttle and utilizing the same assembly and launch facilities at KSC, could satisfy many cargo requirements identified in the CNDB into the next Century. Because of the variety of large payloads and diverse requirements, developing from the detailed studies of the Human Exploration Initiative-Manned Lunar and Mars Missions, various shroud sizes and configurations are now being evaluated including cryogenic oxygen and hydrogen propellant tankers.

A second cargo vehicle study is the joint DOD/NASA Advanced Launch System (ALS) Program. The program has an initial operational capability (IOC) now planned for 200? as depicted in Figure 7. The goal of the ALS program is to minimize the cost per pound of payload delivered to LEO. The ALS concept emphasizes simplicity in design and operation, commonality in propellants, modularity in construction and assembly, a free-standing launch capability, separation of the launch vehicle and the payload interfaces, rapid turnaround, and very high system and mission reliability. The ALS is actually a "family" of vehicles, as shown in (Figure 8), which can be tailored to launch/payload/mission requirements by the addition or deletion of standardized "strap-on" elements. The ALS "family" would provide cargo lift capacity up to possibly 300K pounds.

Assured Crew Return Capability

An Assured Crew Return Vehicle (ACRV) is necessary to provide return of crew from Space Station Freedom in the event of crew medical emergency, a Station Freedom emergency, or the STS being unavailable for an extended period of time. Artists renderings of four distinctly different ACRV concepts are illustrated in Figure 9. At the top left, a Station Crew Return Alternate Module (SCRAM) vehicle is shown; it is based on a simple, aerodynamically stable, seaworthy capsule concept. At the top right, a ballistic reentry configuration based on a Discoverer module is illustrated. Other ballistic concepts include the Apollo derived configuration shown on the lower left. These three are designed for a water landing. At the lower right, a mid-range $^{Lift}/_{Drag}$ lifting-body configuration is shown.

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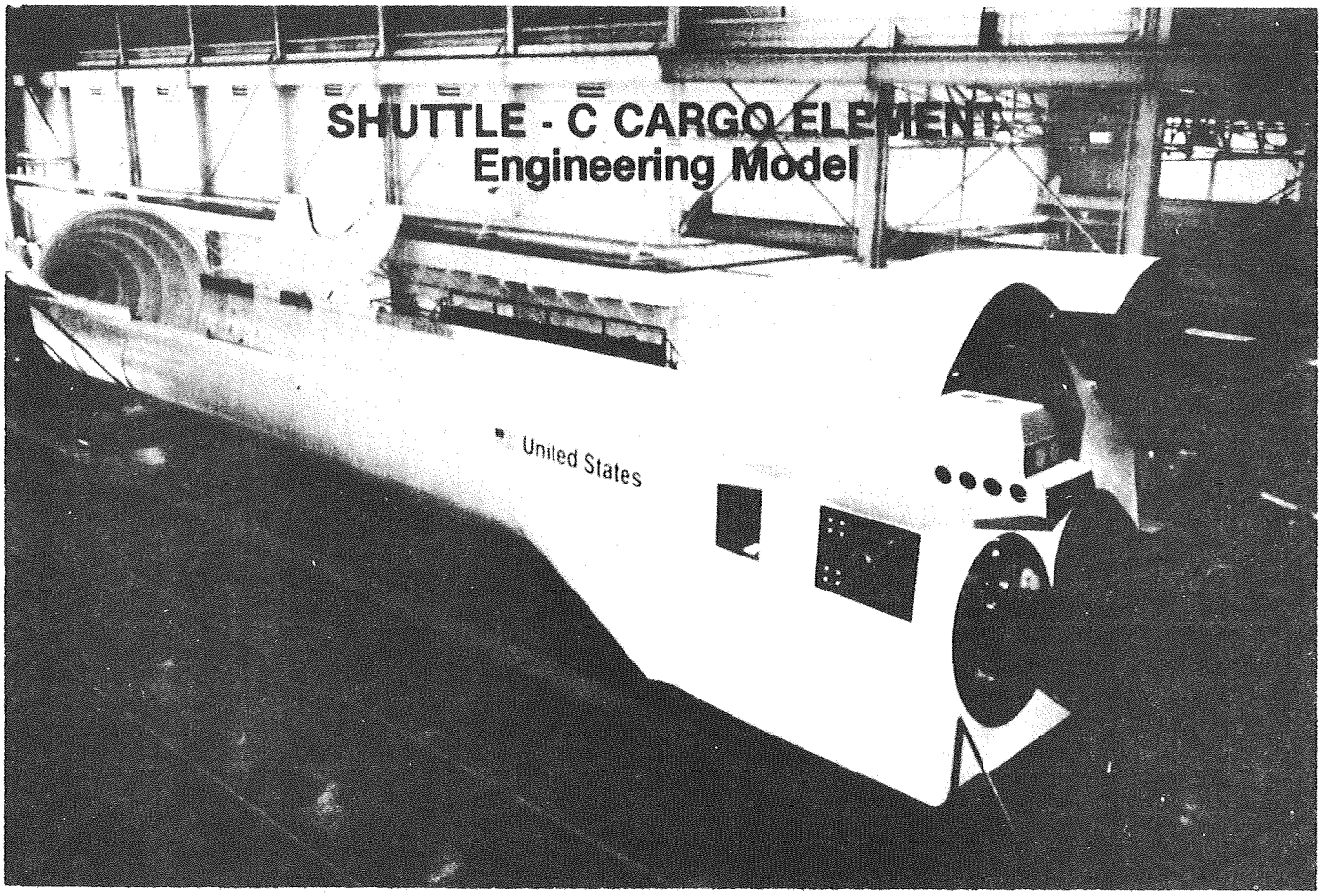


Figure 5. Shuttle C Engineering Model.

Figure 6

SHUTTLE-C SCHEDULE

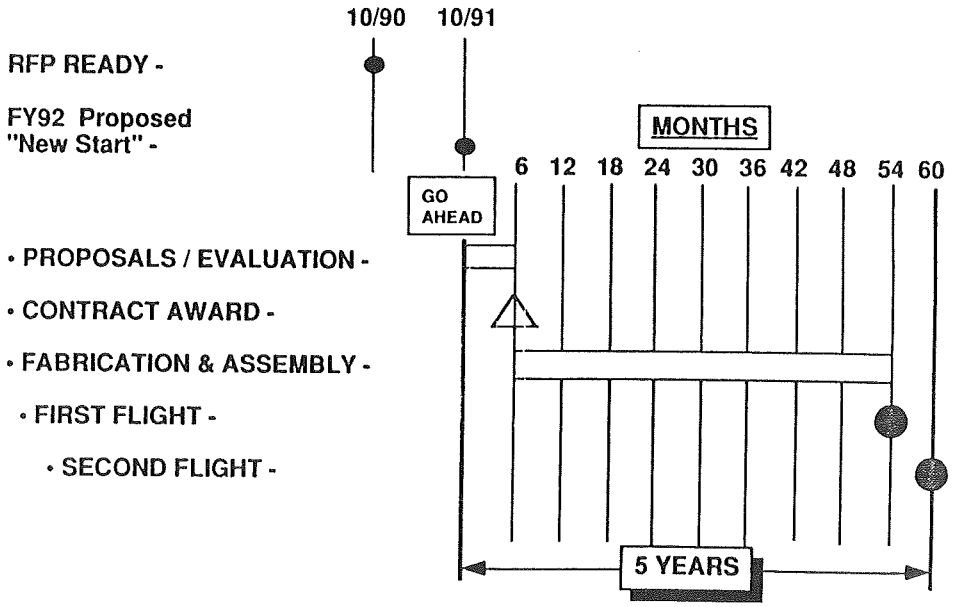


Figure 7

ADVANCED LAUNCH SYSTEM (ALS) SCHEDULE

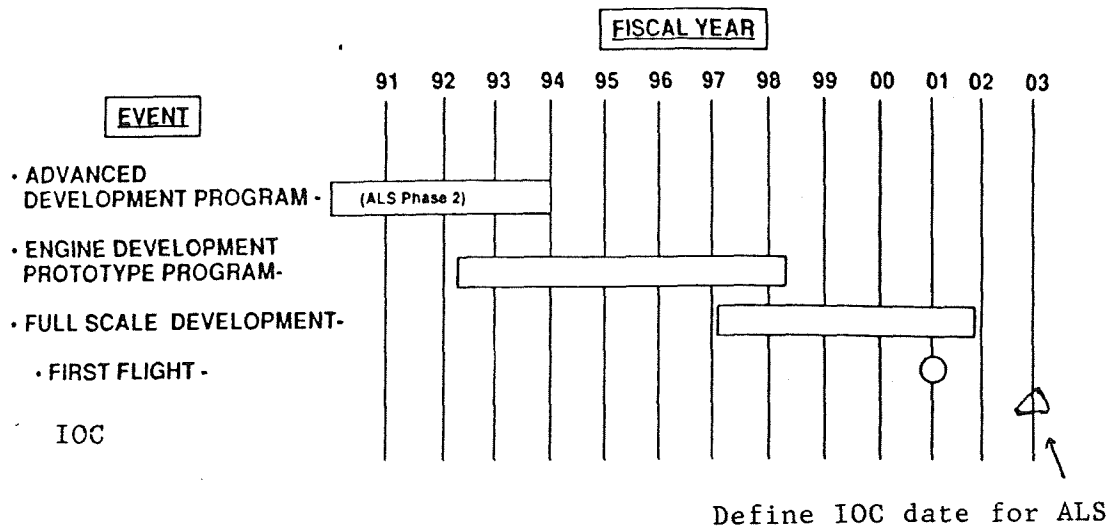
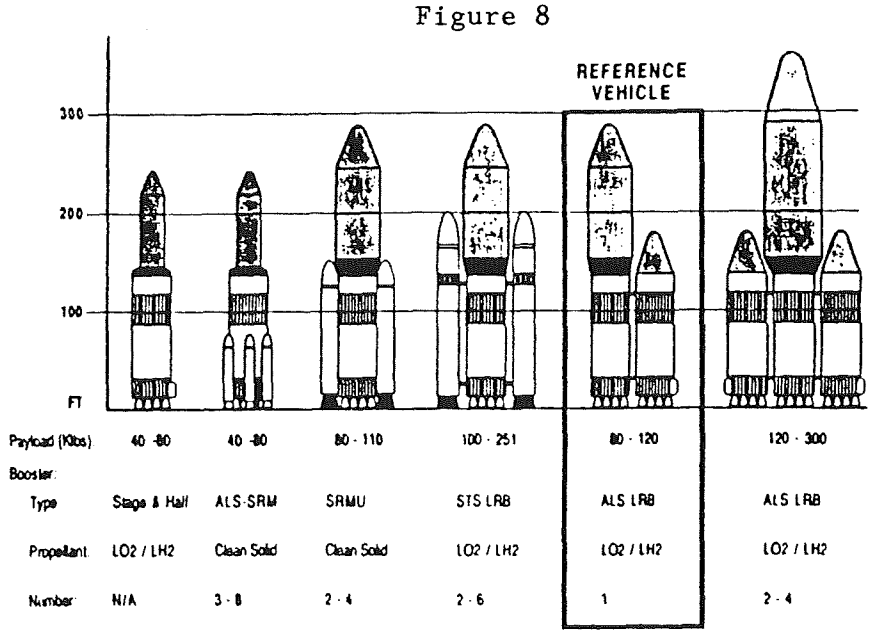


Figure 8



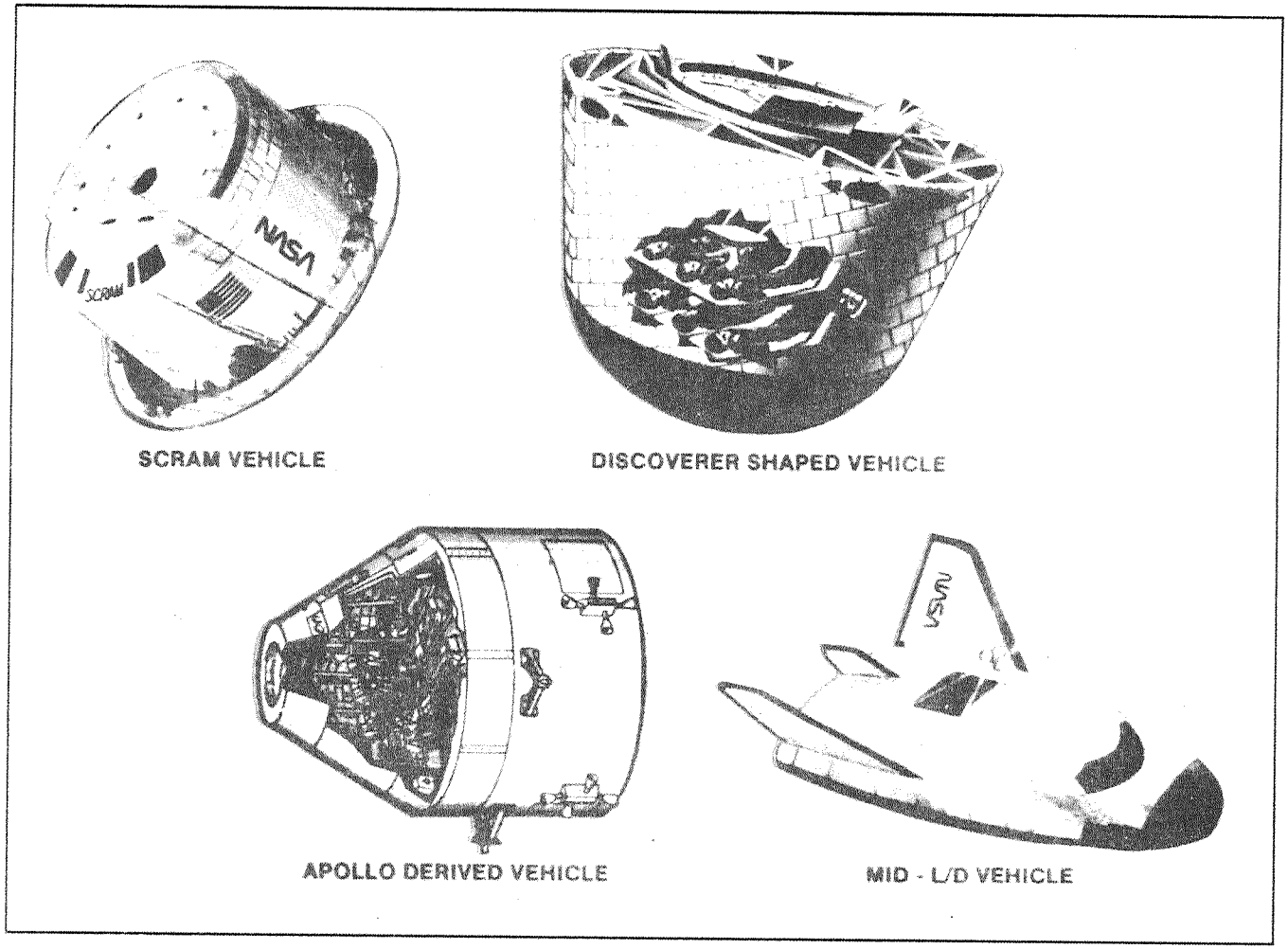


Figure 9. ***ACRV Concepts.***

Competitive assessment of these divergent concepts/configurations including such considerations as crew size, recovery on water or land, and multiple use are under evaluation by two contractor teams.

ACRV in-house studies have been completed, and as shown in the schedule (**Figure 10**), two Phase A-Prime concept and systems definition study contracts are now underway. The teams, one consisting of Lockheed, Boeing and IBM, and the other consisting of Rockwell International, McDonnell-Douglas, TRW and Honeywell have been selected to perform the Phase A-Prime studies. Continuation of the Phase B activities, concentrating on a limited number of vehicle options are planned for the first quarter of FY91.

The Next Manned Transportation System Definition Studies

The existing demand for personnel transport and support of Space Station Freedom extends beyond the projected life span of the existing Shuttle orbiter fleet. Therefore, an integrated space transportation plan for the United States must consider the upgrading or replacement of our manned transportation system. The Shuttle design is now almost 20 years old; new technology is available for a greatly improved design with a significant improvement in performance and cost. The Shuttle was designed as a maximum performance system, is operated near its design limits in almost all areas, and has very little operational margin. The absence of design and operating margin drive the cost of operation and ownership of the Shuttle. The challenge is to define the Next Manned Transportation System (NMTS) design specifications to retain and possibly enhance reliability and safety, yet attain significantly reduced reduced life cycle cost.

The NMTS studies are directed to three very different approaches as shown in **Figure 11**: Shuttle Evolution, Personnel Launch System (PLS), and Advanced Manned Launch System (AMLS). Each approach offers unique design and operational features.

The first, Shuttle Evolution, conceptually illustrated in **Figure 12**, builds on the existing NSTS in an evolutionary, orderly, systematic program to provide specific improvements in performance, cost reductions, and enhanced reliability and safety. Changes could be incorporated in the existing fleet as modifications or retrofits, in the construction of new orbiter vehicles, or in a major redesign of any of the four major STS elements.

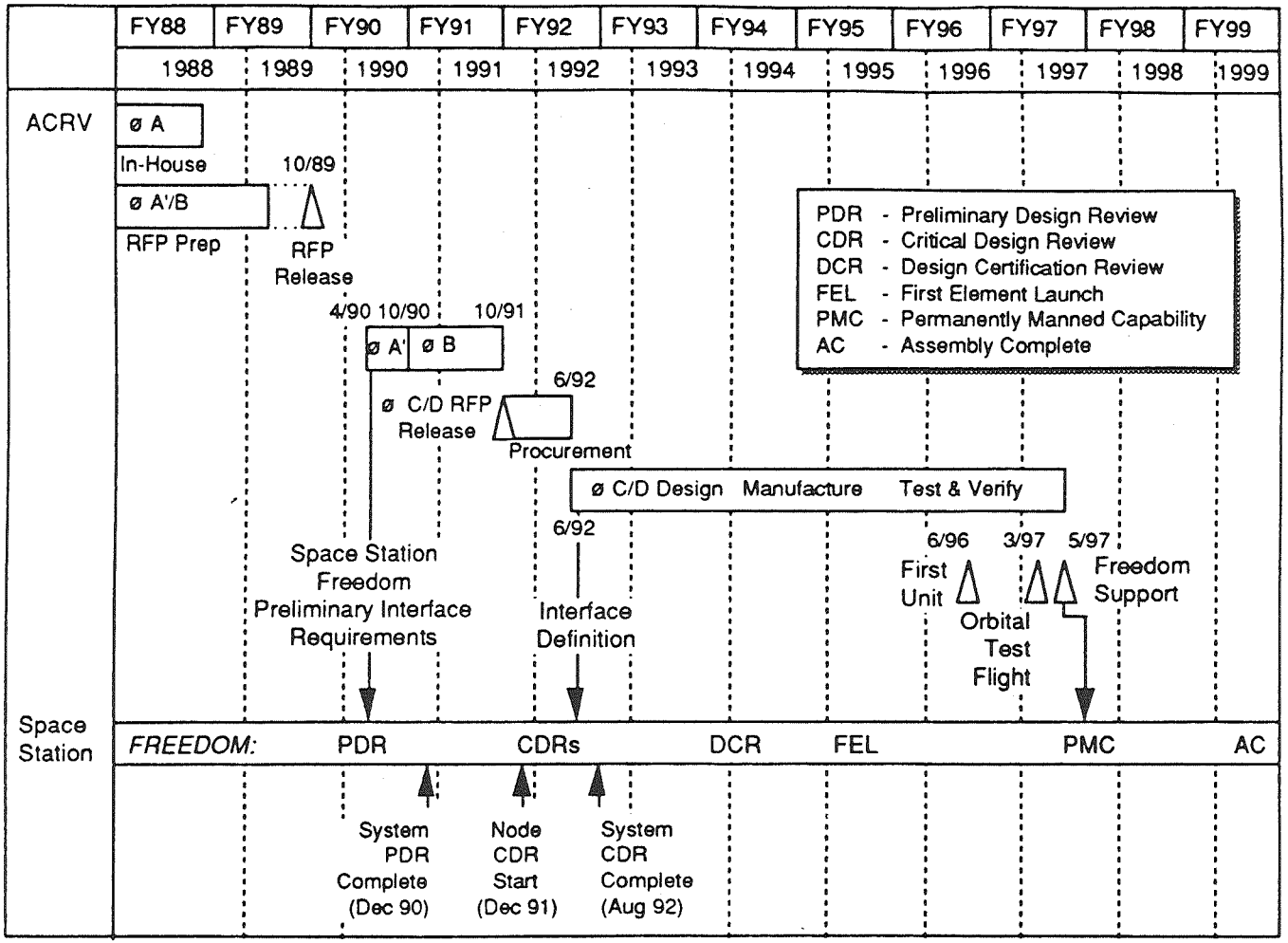


Figure 10. ACRV Schedule.

Figure 11

NMTS CANDIDATE CONCEPTS

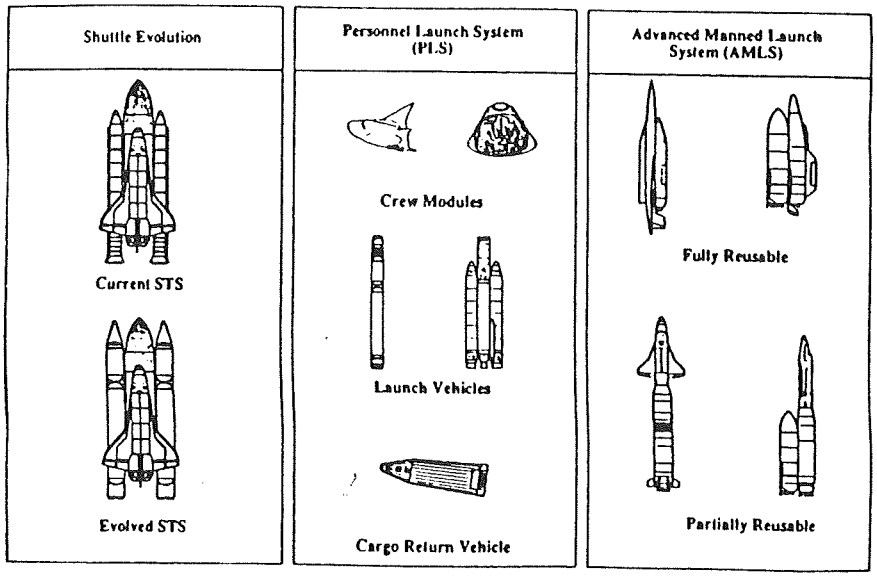
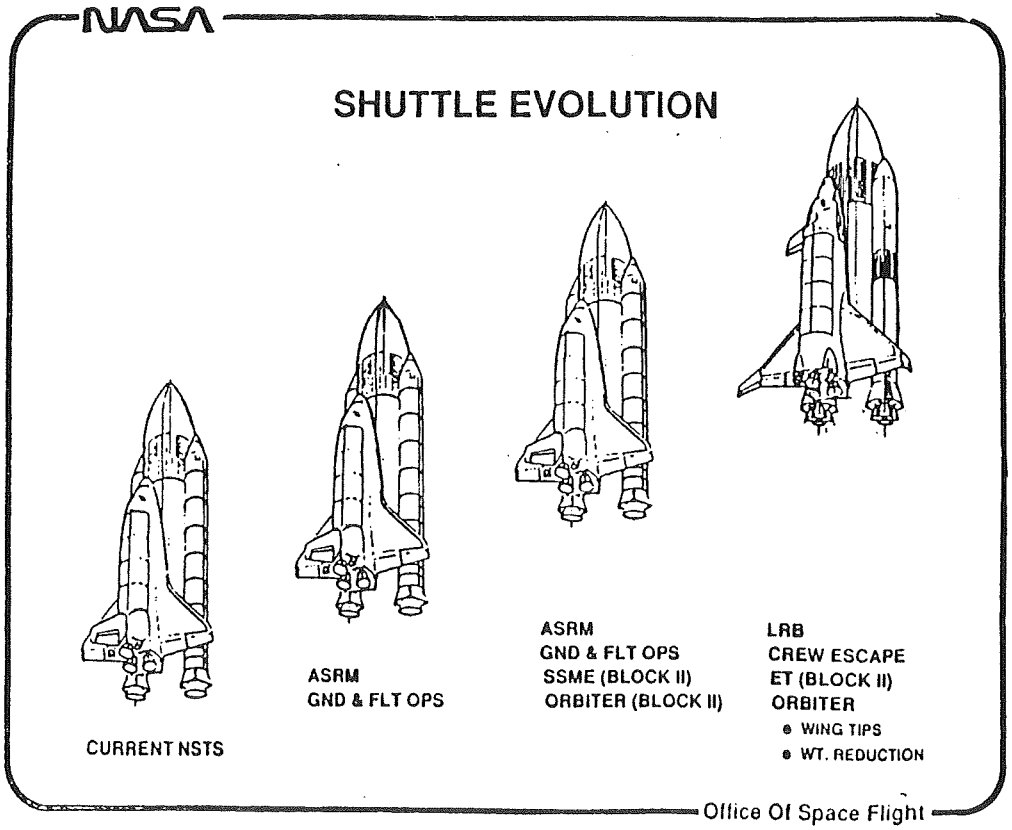


Figure 12



The second, the NMTS option, considers concepts which constrain the vehicle to delivery and recovery of personnel only. This option, in effect, forces virtually all cargo delivery onto Shuttle or dedicated unmanned cargo vehicles. These concepts, shown as the Personnel Launch System (PLS) in **Figure 13**, have the smallest payload requirements. Two study contracts for the conceptual design of the spacecraft capable of transporting a crew of up to10, have been awarded. These studies are to explore whether a ballistic or a lifting body reentry configuration is the preferred concept. The Langley Research Center contract for the lifting body configuration, conceptually illustrated in **Figure 14**, was awarded to Rockwell International Corp. The Johnson Space Center contract for assessment of a ballistic configuration, shown in a launch configuration in **Figure 15**, was awarded to the Boeing Co. These studies both address whether an existing (expendable) launch vehicle or a new launch vehicle is preferred.

The third option being considered in the NMTS assessment is the Advanced Manned Launch System (AMLS), which is a "clean sheet" advanced design to exploit new technologies that become available near the end of the decade. An AMLS, illustrated in **Figure 16**, is conceptualized as a two stage, rocket-powered, fully recoverable, manned, modular launch vehicle system incorporating advanced hypersonic aerodynamics, "hot" structures with advanced high temperature materials, and cryogenic propellants.

The NMTS asessments are now underway to support the NASA out-year budget and planning schedule. Conceptual design studies, followed by a downselection of concepts by the summer of 1991, will support agency decisions on the preferred approach.

Space Transfer Vehicle Definition

NASA is assessing various configurations and design concepts for space transfer vehicles (STV) to deliver geosynchronous payloads, precursor robotic planetary exploration missions and evolution to support human exploration. A conceptualized STV is illustrated in **Figure 17** as a reuseable, space-based, hydrogen/oxygen high performance stage with an aerobraker for either planetary or Earth orbit insertion. The STV would be configured to grow and evolve to provide increased performance capabilities as requirements expand, posssibly evolving from an initially unmanned to a man-rated capability.

The existing Centaur provides a very high level of performance and the RL-10 expander cycle engine is relatively simple and highly reliable. Study activities are underway to explore the potential of upgrading and

THE NEXT MANNED SPACECRAFT

SIMPLE RUGGED PEOPLE CARRIER PATH

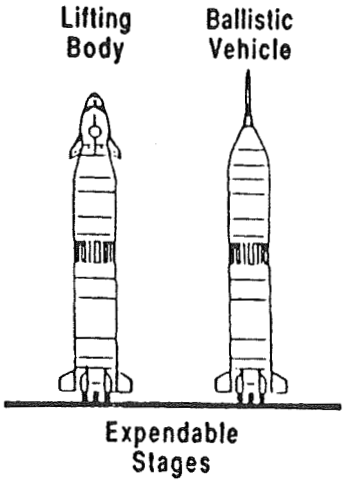


Figure 13

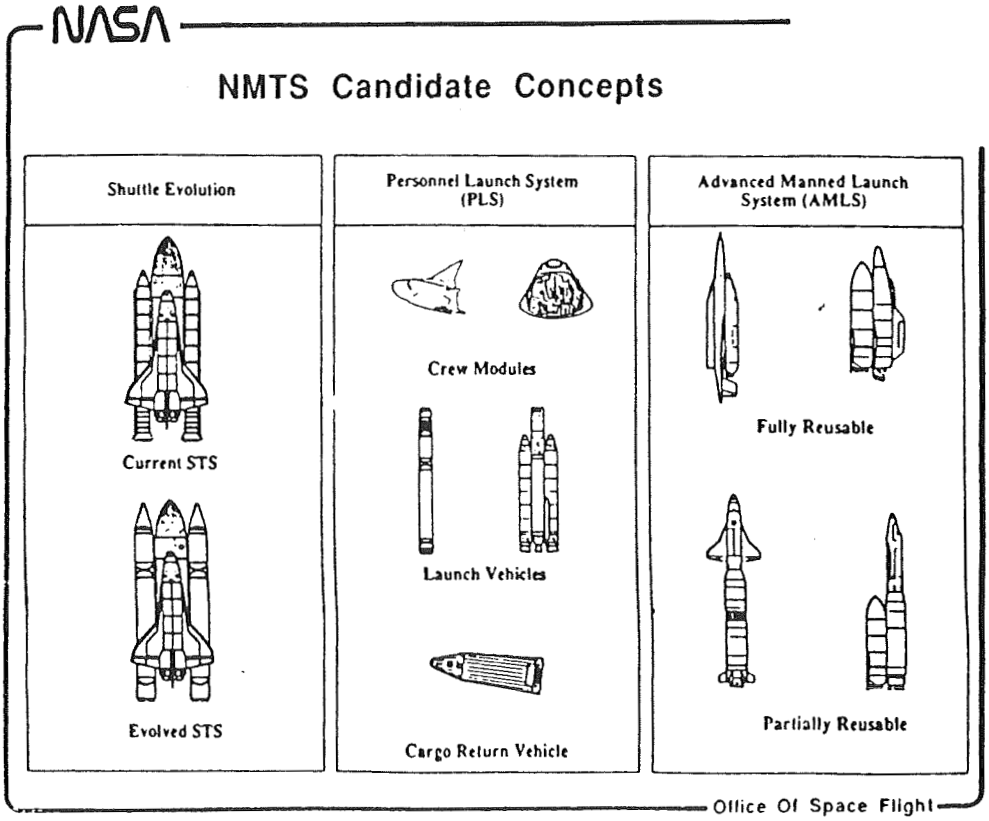


Figure 14

PLS Lifting Body Configuration

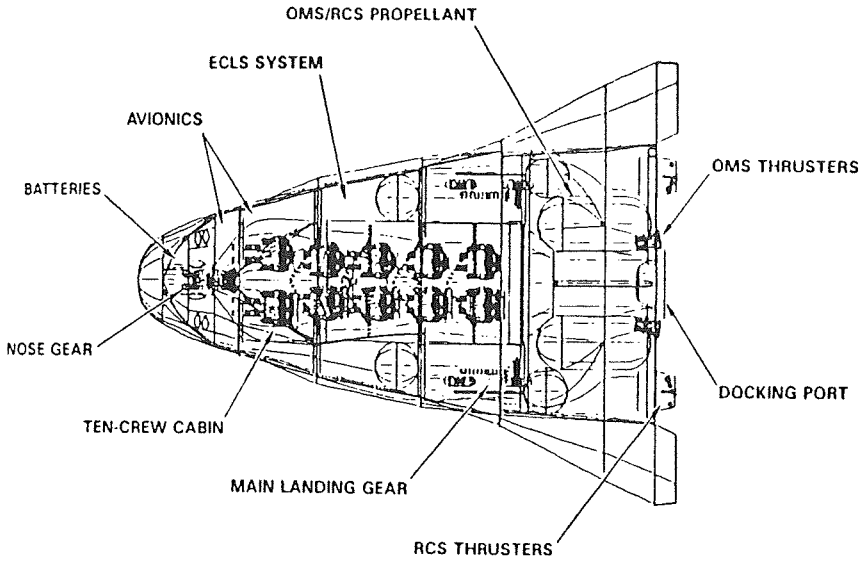


Figure 15

PLS Ballistic Configuration

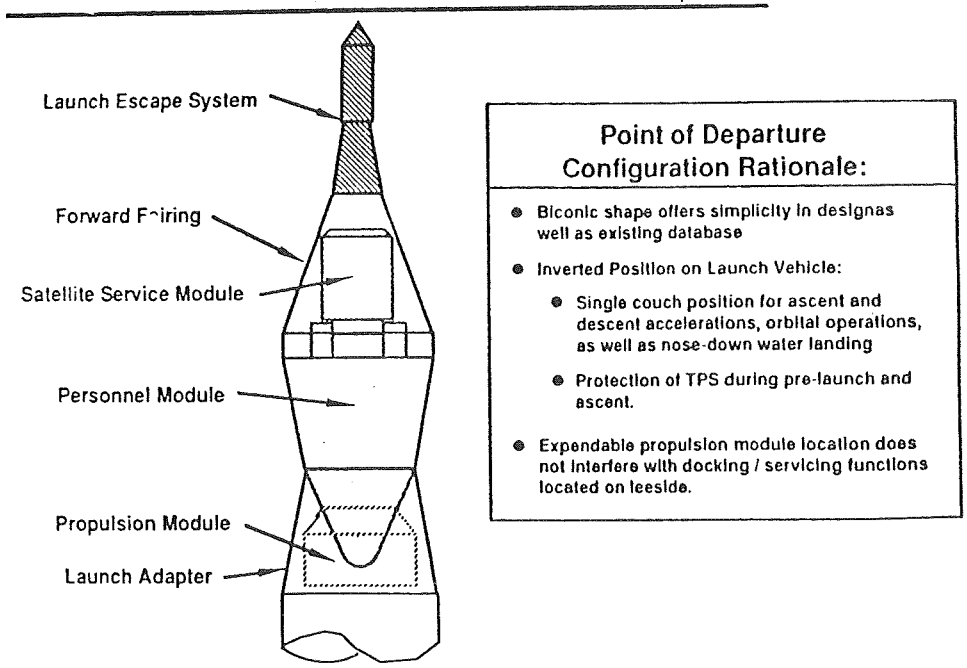
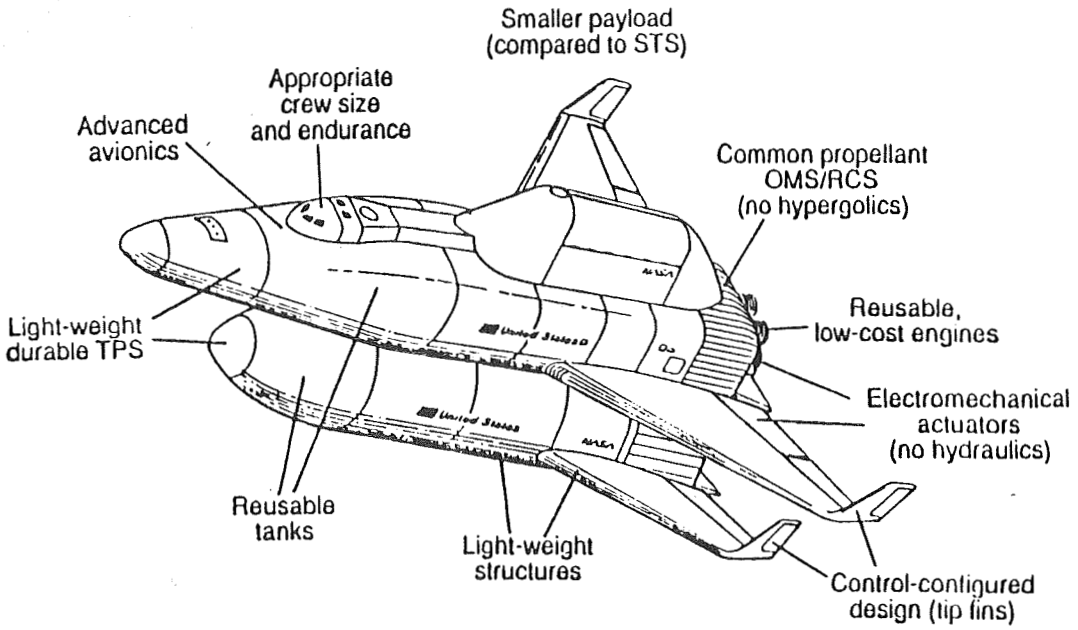


Figure 16

NASA

ADVANCED MANNED LAUNCH SYSTEM

ROLE OF TECHNOLOGY



Office Of Space Flight

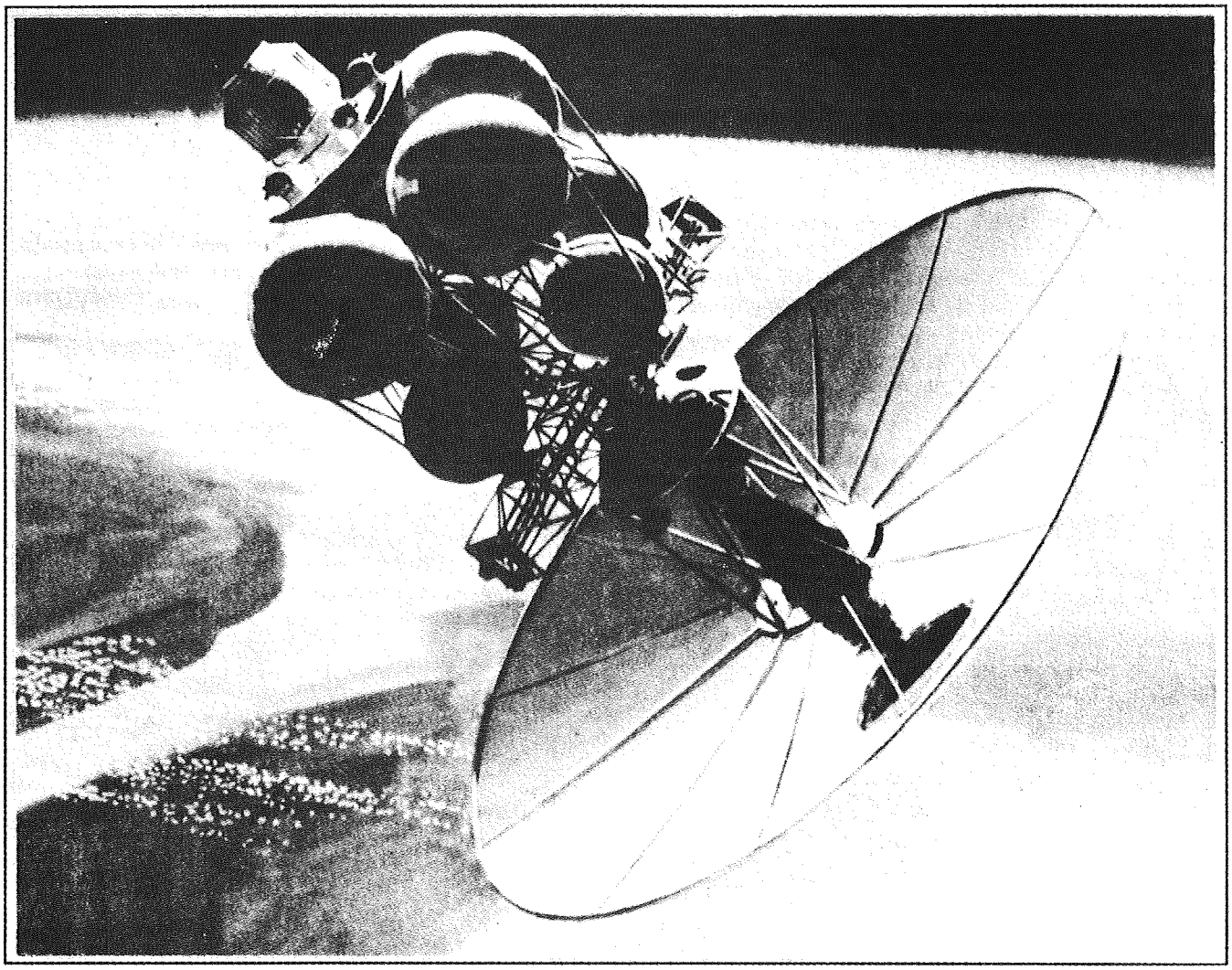


Figure 17. STV Configuration.

re-configuring the Centaur (**Figure 18**) as an unmanned, near term interim upper stage/STV propulsion sub-system.

A concept for a recoverable STV, based on NASA in-house preliminary design studies, is illustrated in **Figure 19**. This concept incorporates an independently recoverable, 12,000 pound propulsion module. The empty tank set could be expended for increased flexibility in operations. Both expendable and recoverable concepts are being evaluated against various high-energy mission requirements. Technology drivers for the STV include aeroassist for atmospheric braking, a new higher performance cryogenic reuseable engine, and in-space cryogenic storage and transfer for reusability. Development and operational cost comparisons and cost prediction models are being developed. Design requirements are being identified for size, thrust levels and operational performance.

Eighteen month, Phase A STV Concept Definition study contracts were awarded to the Boeing Co. and Martin-Marietta Corp. in August of 1989.

Summary

The full flight capabilities of the Shuttle have been reestablished and we are preparing for the deployment of Space Station Freedom. NASA is committed to continuation of the deep space scientific missions and the Earth orbiting systems (EOs) to support "Mission to the Planet Earth. Major planning activities are underway to define the precursor robotic exploration of the solar system and the human exploration missions to the Moon and Mars. These mission planning activities are responsive to the National Goals established by President Bush, to recommendations from the National Commision on Space, to Dr. Rides' report to the Administrator, and to National Space Policy decisions. This guidance clearly defines and establishes major national mission requirements and presents the framework for the evaluation and assessment of long-term space transportation needs.

Long term mission and payload mass requirements have been inventoried in the CNDB. The CNDB provides a framework for the analysis of the launch vehicle requirements and the timeframes when specific launch vehicle and space transfer vehicle capabilities must be available.

System studies in each of the major vehicle classification have been initiated and are underway, each providing necessary information and detail for future decisions. Cargo vehicle studies for Shuttle-C and ALS, provide the increased unmanned lift capacity needed to support expansion of the deep space robotic missions, and the human exploration of the Moon and Mars.

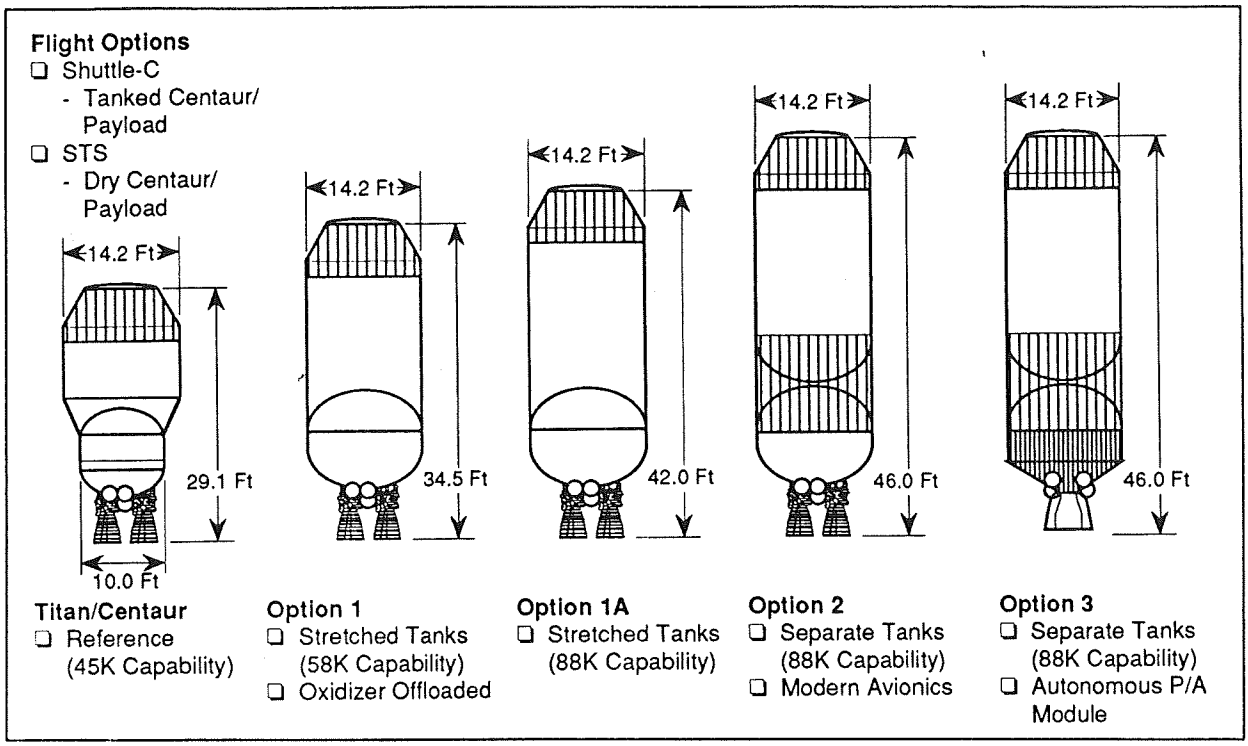


Figure 18. Centaur Upper Stage/STV Planning.

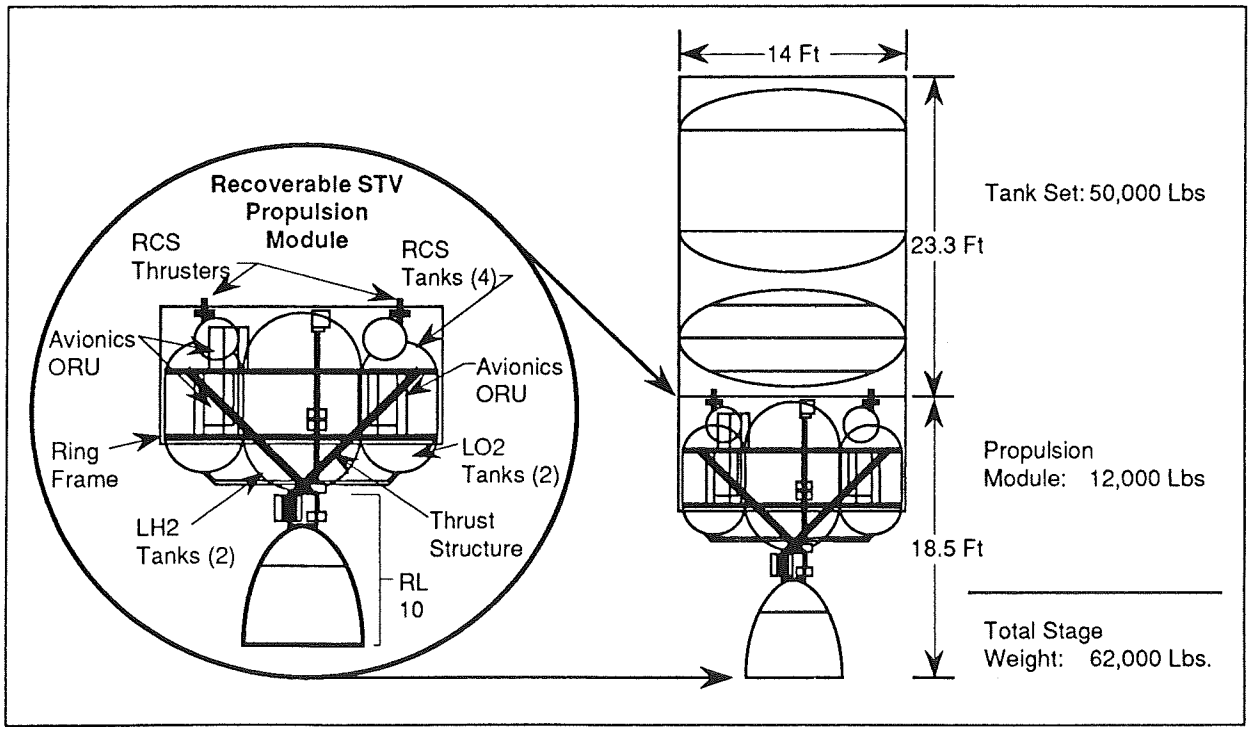


Figure 19. Recoverable STV Propulsion Concept.

The NMTS studies are directed to the definition of options for manned flight beyond the current Shuttle capabilities. Each of the three NMTS studies underway are unique: Shuttle Evolution adding technological improvements and building upon assets and capabilities inherent in the Shuttle, the PLS, directed exclusively to personnel/crew launch and recovery, and the AMLS representing a next generation capability based on advanced technology. These Next Manned Transportation System studies, are in support of the decision, (planned for 1992), on how the United States manned vehicle development program should proceed.

The ACRC studies are on a schedule to provide a necessary crew return capability for the Space Station Freedom-Permanently Manned Capability (PMC) in the summer of 1997.

The STV activities will define space-based, aerobraking, cryogenic, vehicle concepts that will permit multiple reuse from LEO and will evolve over time to support expanded unmanned and manned exploration missions. Space Station Freedom will function as a node for STV space-basing, on-orbit servicing, and resupply.

A broad and diverse range of future requirements have been identified. Lead times for transportation systems are very long and future needs must be anticipated well in advance. The challenge is to satisfy these requirements, in a time phased sequence, to assure that both lift capacity and operational capabilities are available when needed. The studies and programs described are in place and are structured to support the definition of an integrated advanced transportation system for the United States. Over the next several years we must define an advanced transportation system that can sustain the evolutionary manned space flight program envisioned by the President and the American public. These systems will form the basis for a space transportation system that will satisfy projected mission and traffic demand well into the next century.

References

1. President Ronald Reagan, Space Station decision, State of the Union Message to a joint session of the U.S. Congress, January 24, 1984.

2 The National Commission on Space, "Pioneering the Space Frontier", Bantam Books, May 1986.

3. Ride, Dr. Sally K, Leadership and America's Future in Space, A Report to the NASA Administrator, August 1987.

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PRESENTATION 1.1.3

MAINTAINING TECHNICAL EXCELLENCE

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Maintaining Technical Excellence Requires a National Plan

T. F. Davidson

For Presentation at the NASA 1990
Symposium on Space Transportation
Propulsion System Technology,
Pennsylvania State University,
25–29 June 1990

MAINTAINING TECHNICAL EXCELLENCE REQUIRES A NATIONAL PLAN

by T. F. Davidson

"Where there is no vision, the people perish." (Proverbs 29:18)

Rocket propulsion is the cornerstone of every space transportation system. Since the late 1950s, the United States has been the undisputed world rocket propulsion leader. However, the technical excellence and technology base that earned us such a reputation have been eroding. Foreign competition now threatens to overtake this country early in the next century.

In the 21st century, rocket propulsion will become an increasingly important part of international trade. Without a change in national policy and a commitment to a strong, continuing, broad-based rocket propulsion technology program, the United States' position will continue to erode, possibly to a point of no return. Without a commitment to technical excellence we will fail!

The Global Picture

- National position eroding
- Foreign competition increasing
- National technology imperative needed
- National commitment needed
- Commitment to technical excellence needed
- National plan needed

This was the picture visualized by the Aerospace Industries Association (AIA) in 1987, and this is why it selected rocket propulsion as one of their 10 key technologies for the year 2000 (Figure 1).

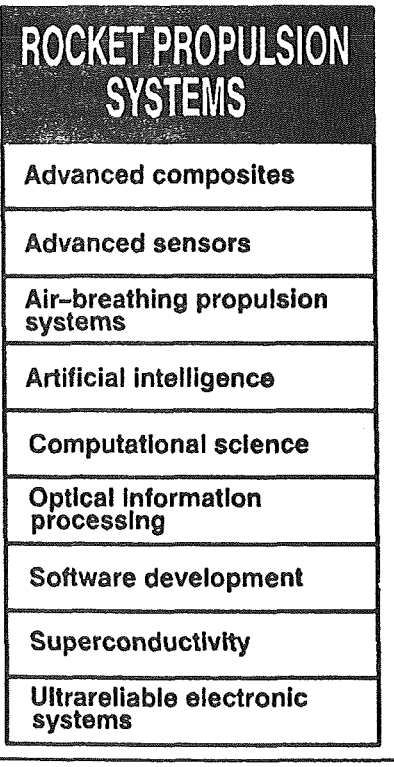


Figure 1. AIA Key Technologies for the 1990s

To meet the challenge, AIA established a rocket propulsion committee (which I had the privilege of chairing until my retirement earlier this year) to develop the *National Rocket Propulsion Strategic Plan*. Developing such a plan required a broad spectrum of experience and disciplines. The *Strategic Plan* team needed the participation of industry, Government and academia. The list below tends to understate the number of participating organizations, since in many cases multiple divisions and centers participated.

The *Strategic Plan* Team

Industry	Government
Aerojet	Department of Defense
The Aerospace Corporation	Department of Energy
Aerospace Industries Association	NASA
Atlantic Research	U.S. Air Force
Boeing Company	U.S. Army
General Dynamics Corporation	U.S. Navy
Hercules Inc.	
LTV Corporation	**Academia**
The Marquardt Company	Brigham Young University
Martin Marietta Corporation	California Institute of Technology
Rockwell International Corporation	Georgia Institute of Technology
Science Applications International Corp.	Pennsylvania State University
Sverdrup Technology	Purdue University
Thiokol Corporation	University of Akron
TRW, Inc.	University of Alabama
United Technologies Corporation	University of Delaware
Wyle Laboratories	University of Illinois
	University of Texas

Six NASA organizations participated in developing and commenting on *Strategic Plan* drafts:

- NASA Headquarters
- Langley Research Center
- Lewis Research Center
- Marshall Space Flight Center
- Stennis Space Center
- Jet Propulsion Laboratory

All told, from March 1988 to the present, over 50 organizations and 200 people have participated in developing the *Strategic Plan*. Such participation was necessary to ensure a national consensus. It took basically two years, 10 meetings and a great deal of dedicated, hard work to reach a plan draft that was ready for comprehensive, detailed independent review. The review was accomplished in two phases. In the first phase (October 1989), draft copies of the plan were sent to 137 organizations for review and comment. These included industry, Government and university organizations, as well as selected AIA and American Institute of Aeronautics and Astronautics (AIAA) technical committees. The October review yielded some 250 pages of comments. In the second phase, a symposium was held in Washington on 15 February to brief the plan. The symposium was sponsored by the National Center for Advanced Technologies (NCAT), a nonprofit educational foundation established by AIA to coordinate and integrate its Key Technologies effort. Two hundred attendees participated in the symposium. They were briefed, given copies of a revised plan draft and invited to submit their comments to NCAT for incorporation into the final plan. At the symposium, 60 questions were raised, recorded and answered in writing.

Plan Chronology

- Team generates plan 1988–1989
- First independent review October 1989
- NCAT symposium February 1990
- Second independent review March 1990
- Issue plan July 1990

The plan was redrafted in May and will be distributed in July. The plan provides, if followed, a means for the U.S. to maintain technical excellence and world leadership in rocket propulsion. To implement the *National Rocket Propulsion Strategic Plan* is to invest in the social, economic and technological futures of America. It is the way to maintain TECHNICAL EXCELLENCE in rocket propulsion (Figure 2).

The *National Rocket Propulsion Strategic Plan* is a roadmap of technologies and strategies designed to maintain America's technical excellence and global competitive posture.

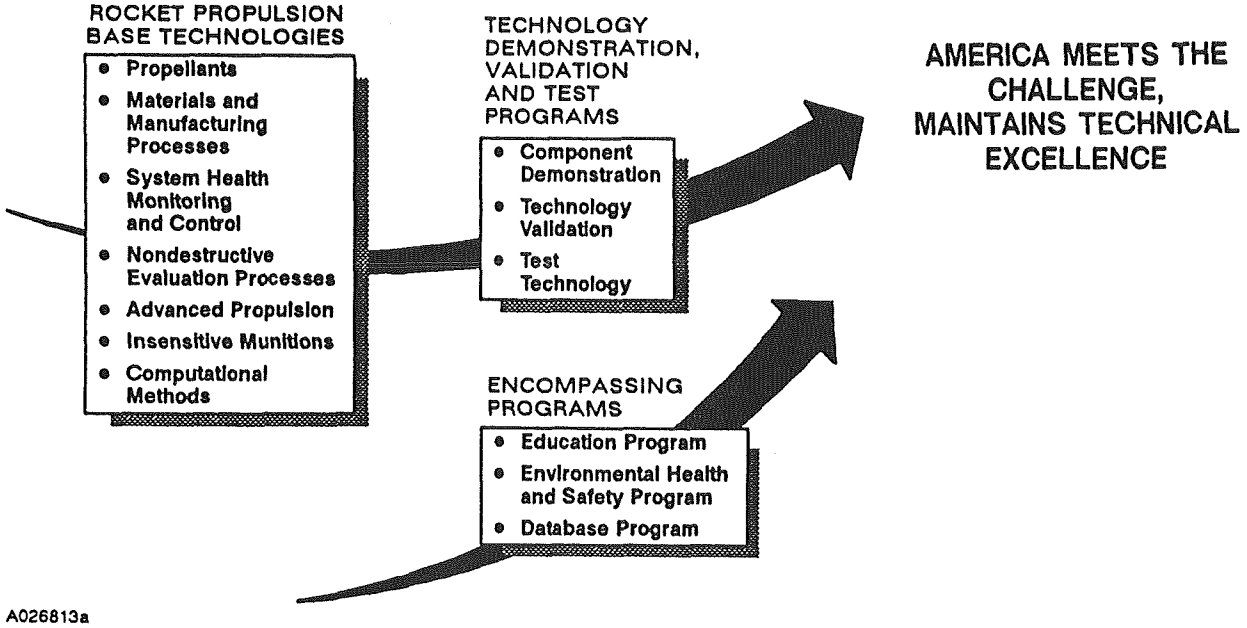


Figure 2. The Plan

I encourage you to read the plan.* In my opinion, this plan represents a national consensus of what needs to be done to maintain technical excellence in the 21st century. I would like to take this opportunity to express my appreciation to the over 200 people who helped prepare the plan and the approximately 600 people who reviewed it.

* *Distribution is authorized only to U.S. Government agencies and their contractors. Attendees at the February 1990 symposium will automatically receive copies. Additional copies ($100 each/$50 for universities and libraries) may be obtained by contacting Mr. R. H. Hartke, National Center for Advanced Technologies, 1250 Eye Street N.W., Washington, D.C. 20005*

The following is a synopsis of the *Strategic Plan*'s major parts:

- The challenge
- Base technology programs
- Technology demonstration, validation and test programs
- Encompassing programs
- Implementation

The executive summary presents the basic challenge and explains why maintaining rocket propulsion leadership must be a national technology imperative, the theme of the February NCAT symposium. The *Strategic Plan* lays the basis for upgrading existing propulsion systems and a firm base for future full-scale development, production and operation of rocket propulsion systems for space, defense and commercial applications.

The challenge simply stated is: National supremacy is fading, foreign competition is real and increasing, current full-scale development cycles take too long and cost too much and technology support has been declining.

Table 1 shows the growth of foreign competition and capability since 1968. In many areas of both liquid and solid rocket propulsion technology, foreign competition has already overtaken the United States. Four examples come to mind: 1) the French are ahead of us in carbon/carbon composites and a basic understanding of electrostatic discharge, 2) the British are ahead on plume tailoring fundamentals and 3) the Japanese are ahead in the use of ceramic bearings. The National Science Foundation's (NSF) evaluation of Japanese liquid rocket technology and plans last fall left little doubt that Japan intends to have a completely autonomous rocket and launch capability by the end of this decade.

Table 1. The Reality of Foreign Competition

	1968		1990	
	Liquid	Solid	Liquid	Solid
1. Argentina				X
2. Australia				X
3. Brazil				X
4. Canada		X		X
5. China			X	X
6. Egypt				X
7. France	X	X	X	X
8. Great Britain	X	X	X	X
9. Greece				X
10. India			X	X
11. Israel				X
12. Italy			X	X
13. Japan	X	X	X	X
14. Norway				X
15. South Africa				X
16. South Korea				X
17. Sweden				X
18. Switzerland				X
19. Taiwan				X
20. Turkey				X
21. U.S.S.R.	X	X	X	X
22. West Germany	X		X	X
23. Yugoslavia	—	—	—	X
Total	5	5	8	23

X—Viable foreign competition

Figure 3 shows a typical development schedule for a new propulsion system over the last 20 years and what the key goals of the plan are: increased reliability, lower risk, shorter time and less costly development.

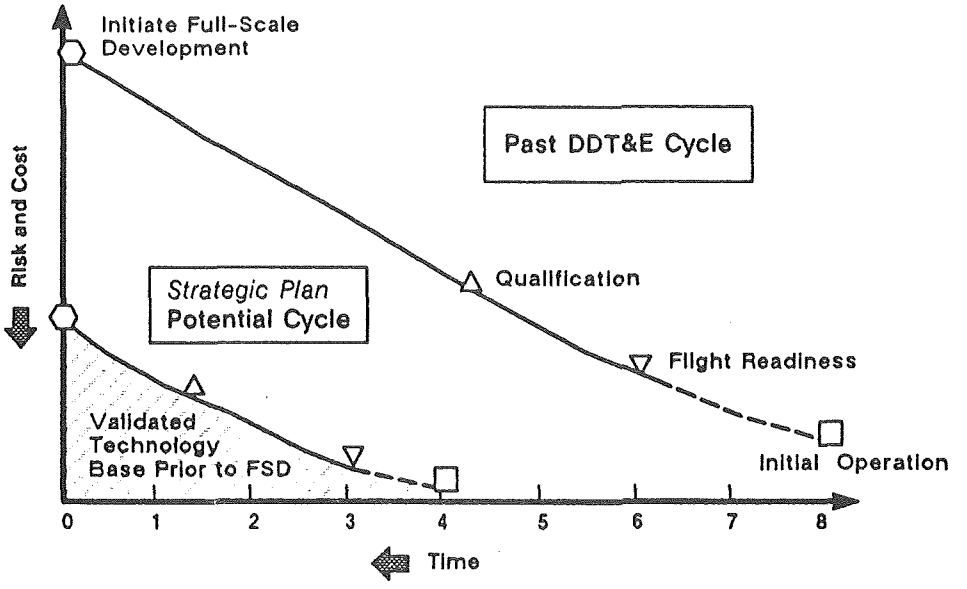


Figure 3. A Strong Technology Base Will Reduce Risk, Cost and Schedule

During the 1950s and 1960s, technology support, as a percent of total rocket propulsion expenditures, averaged approximately 10 percent. The results of such technology investment were applied to many propulsion systems, e.g., Scout, Apollo and Space Transportation System (STS). In the early 1970s, technology support declined rapidly and has never regained the position it enjoyed earlier. We have coined this era the Rocket Technology Drought (Figure 4). The drought, which applies equally to all Department of Defense (DoD) systems, was a contributor to several space propulsion failures.

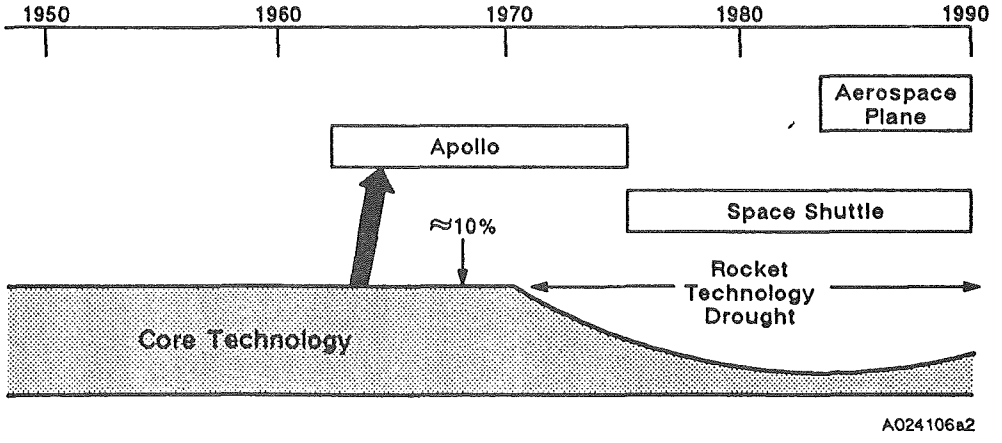


Figure 4. The Technology Drought

In the 1990s and beyond, reliability, safety (which includes health and environmental concerns) and cost reduction must be accepted as technical goals on the same basis as performance goals have been in the past. Rocket propulsion must be a national technology imperative. Figure 5 sums up the problem, the challenge and the solution.

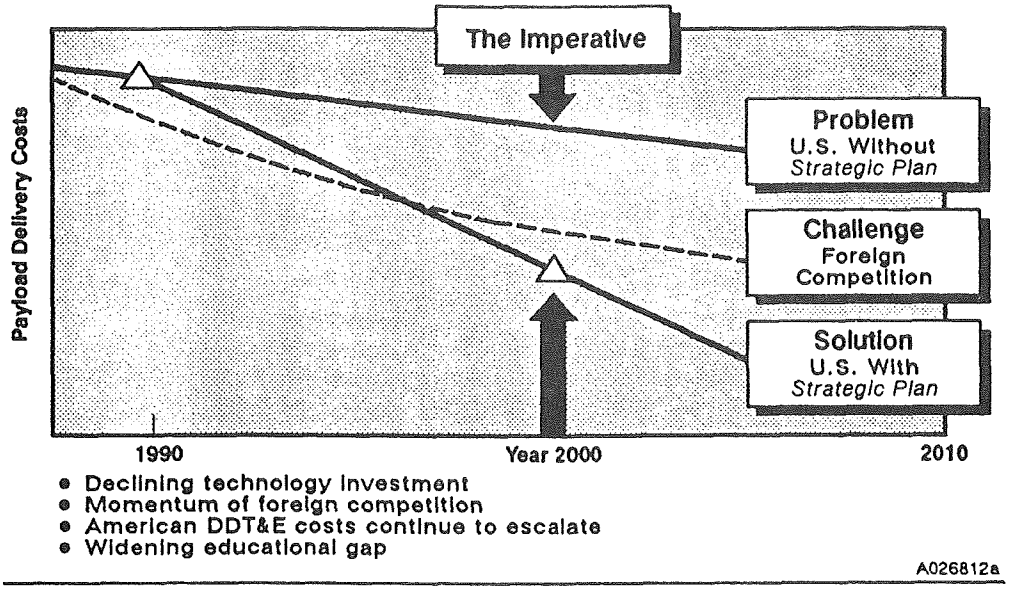


Figure 5. Why an Imperative?

The basic technology improvement areas are shown in Figure 6. The *Strategic Plan* was developed to support perceived military and space objectives and schedules. Space objectives through the year 2020 are shown in Figure 7. Rocket propulsion technology must be developed and validated during the 1990s to support future needs because of the severe environments that are unique to propulsion technology.

Why Propulsion Technologies Must Be Developed Early

- Propulsion systems have the most severe:
 - Forces
 - Pressures
 - Temperatures
 - Heat fluxes
 - Material environments
 - Energy densities
 - Vibration levels
- 21st century improvements, therefore, must start in the 1990s

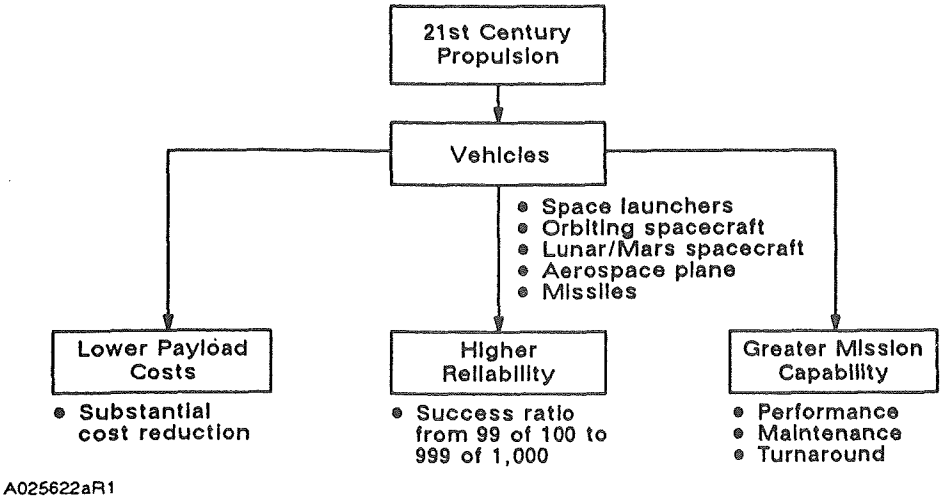


Figure 6. Improvements Are Needed for U.S. Aerospace Competitiveness

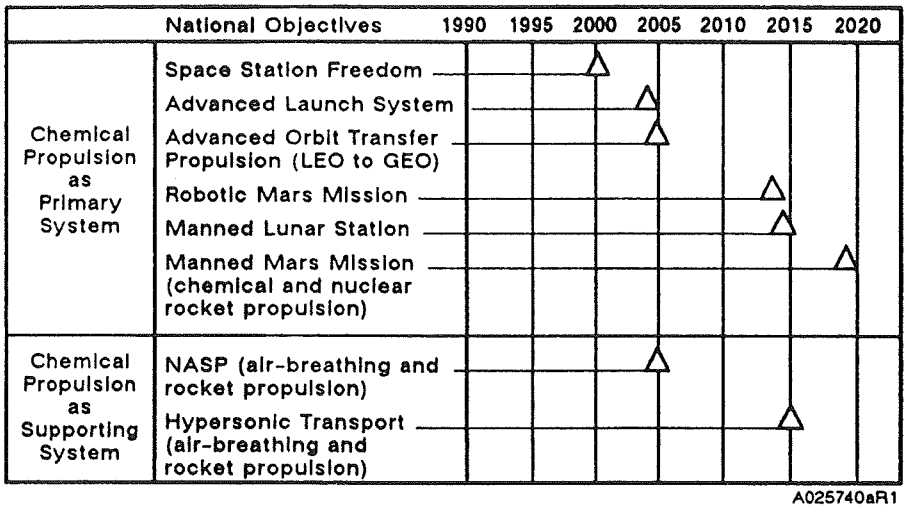


Figure 7. The Plan Supports Future National Space Objectives

Figure 8 charts the technical sections and basic phases of the plan. Under Base Technologies, objectives, overall approach, schedule and costs have been defined for chemical rocket propulsion (solid rocket, liquid rocket, hybrid rocket and advanced concepts) in the following areas, which encompass 258 individual programs:

- Propellants
- Materials and manufacturing processes
- Health monitoring and control
- Nondestructive evaluation
- Computational methods
- Insensitive munitions
- Advanced propulsion concepts

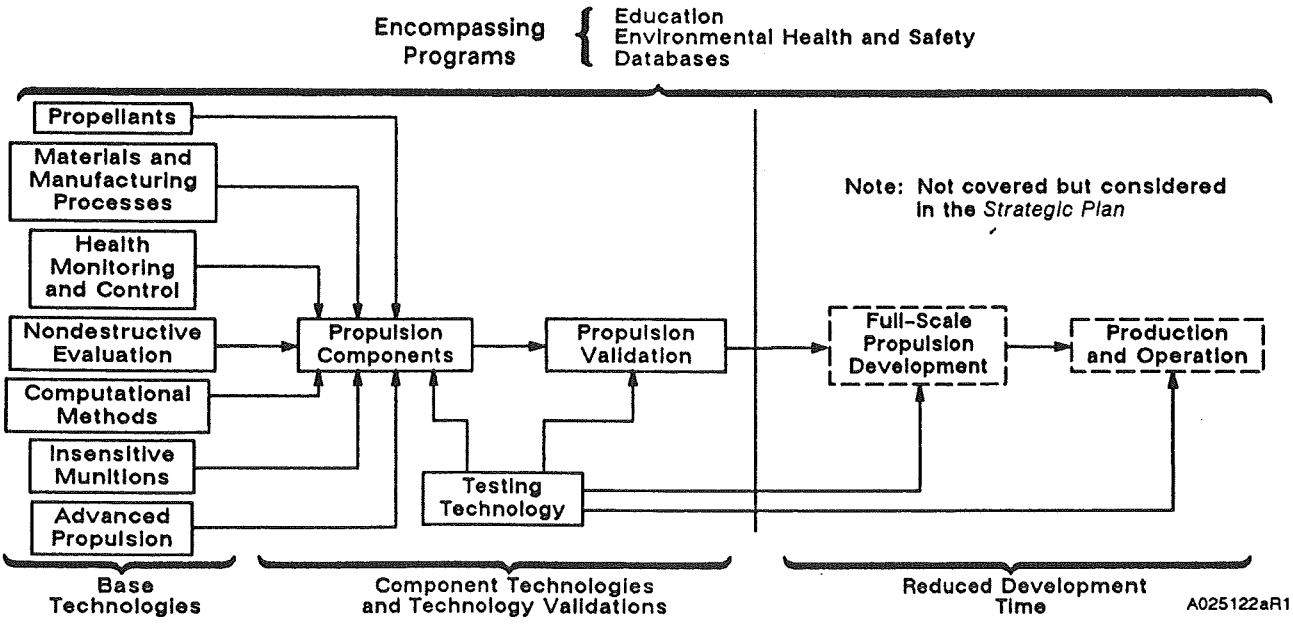


Figure 8. The Plan Addresses the Key Areas in Rocket Propulsion Technology

Base technology will flow into propulsion component development and demonstration, then into the prototype system validation phase, as illustrated in Figure 9. There is also a need to first develop, then validate new testing techniques, instrumentation, diagnostic approaches and automated expert test data analysis systems.

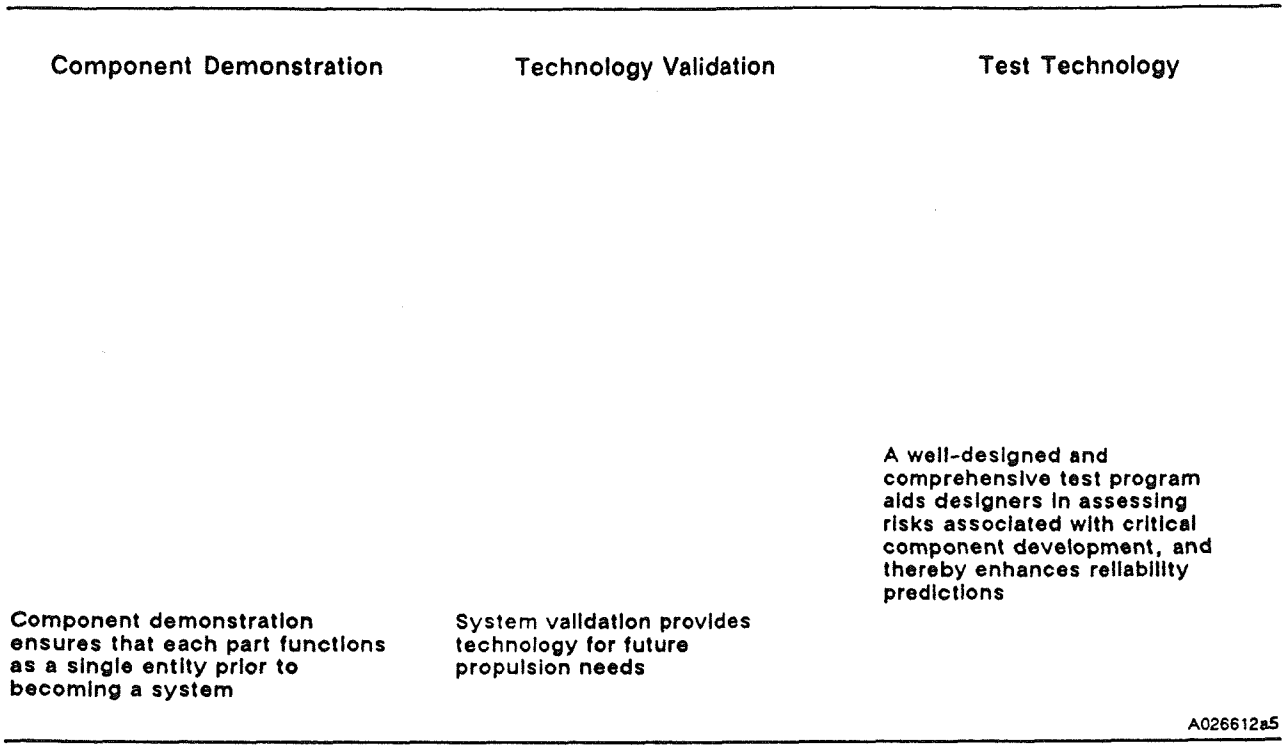


Figure 9. Technology Validation Process

The third area covered in the plan comprises Encompassing Programs. Encompassing programs are needed to ensure technical excellence. They fall into three categories:

- Databases
- Environmental health and safety
- Education

Only those programs needed to support rocket propulsion technology are presented, but in most cases these should fit into required larger, across-the-board national efforts.

Accurate storage, retrieval and rapid dispersion of data, as shown below, are essential for the future of rocket propulsion technology.

Database Elements

Database Management	Materials Properties	Design/Processing
Select and specify hardware and software for centralized data management and maintenance	Provide standardized material properties for use in probabilistic design techniques	Maintain documentation of analytical methods and lessons learned. Databases will affect the formulation of industry-wide standards

Environmental health and safety (EHS) impacts must be considered and minimized in all future propulsion efforts. Industry-wide standards for risk assessment and management of design and process characteristics that address human health and preservation of the environment must be developed.

- Establish aerospace safety and environmental center
- Create environmental working group
- Identify new hazards and failure prediction and detection technologies
- Improve computer simulation and modeling techniques

An area of increasing concern, education is a prerequisite for the U.S. to maintain technical excellence and global competitiveness. The problem is summarized below:

- Education
- Aerospace needs and industry will grow in 21st century
- U.S. rocket scientists and engineers retiring
- Must attract students to technical fields
- Must train students for technical fields
- Rocket community must do its share

Figure 10 illustrates the types of programs we think necessary. Efforts such as those currently being undertaken at the Penn State Space Propulsion Engineering Research Center are an excellent example of what needs to be done. These should be expanded whenever feasible.

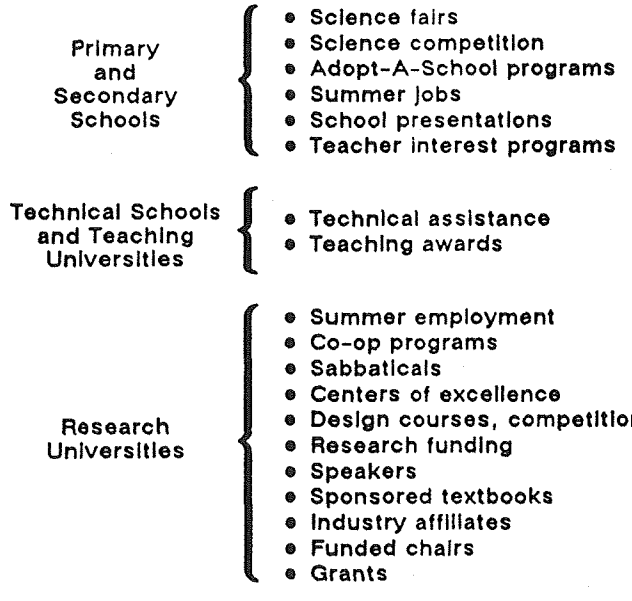


Figure 10. Education Programs

If implemented, the plan will provide a host of technical payoffs to the country, some of which are shown in Figure 11.

The 303 programs detailed in the plan will cost approximately $5.3 billion (a significant financial investment) over the next 10 years (Table 2).

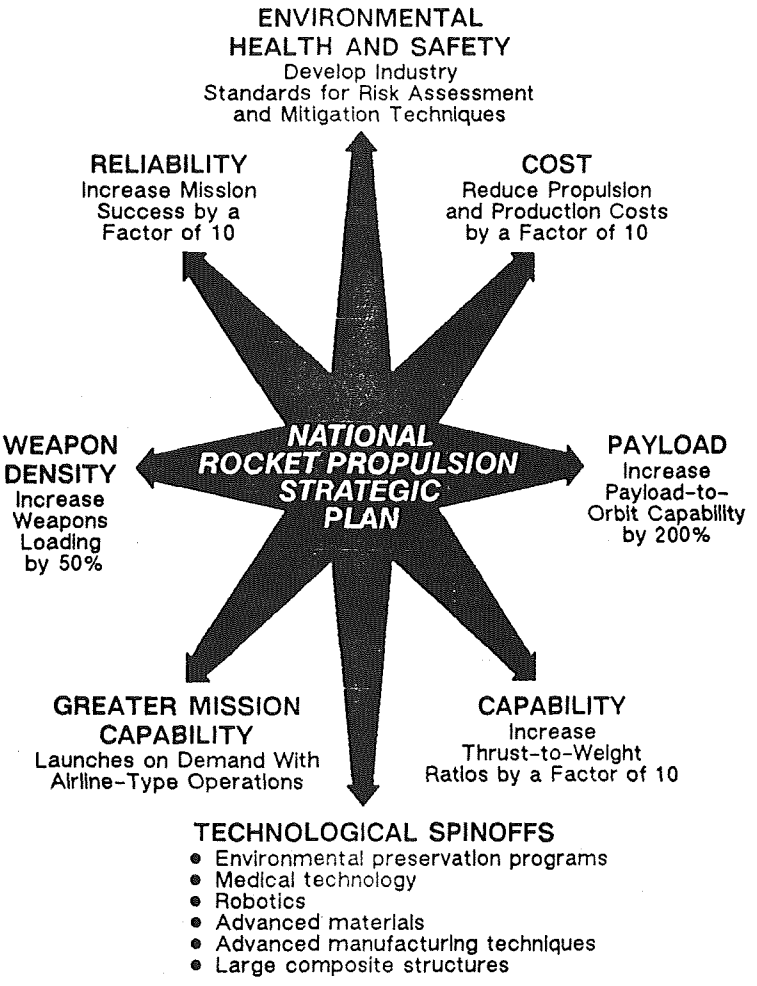


Figure 11. Technical Payoffs

Table 2. Program Summary

	Cost of Programs ($M)	Number of Programs
Propellants	538.20	60
Materials and Manufacturing Processes	561.20	63
Computational Methods	275.40	32
Health Monitoring and Control	125.00	14
Nondestructive Evaluation	144.00	10
Insensitive Munitions	159.10	17
Environmental Health and Safety	145.30	7
Liquid Rocket Components	325.00	27
Solid Rocket Components	273.70	18
Advanced Propulsion Concepts	457.50	10
Test Technology	333.50	11
Propulsion Validations	1,821.00	23
Databases	106.60	11
Total	5,265.90	303
Education		19

On an average annual basis ($527 million/year), this represents an increase of 160 percent over FY89 levels ($202 million estimated). Such an increase will require a national commitment and "ramp up" of approximately 18 percent per year from FY89 year levels through the mid-1990s. Of the total funding, approximately 30 percent should come from industry (IR&D, capital expenditures, etc.) and the remainder from the Government (DoD and NASA).

Figures 12, 13, and 14 present projected annual costs, benefit distribution by mission (including DoD) and benefit distribution by end item user, respectively.

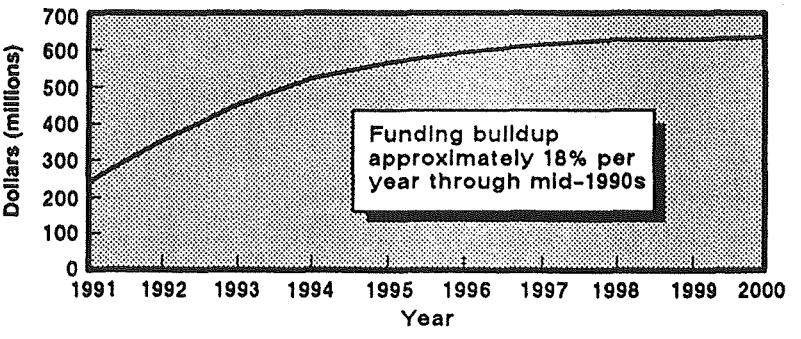
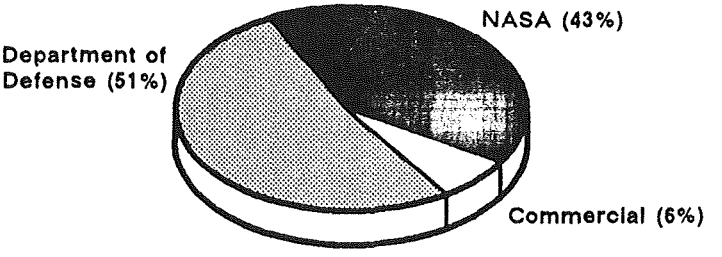
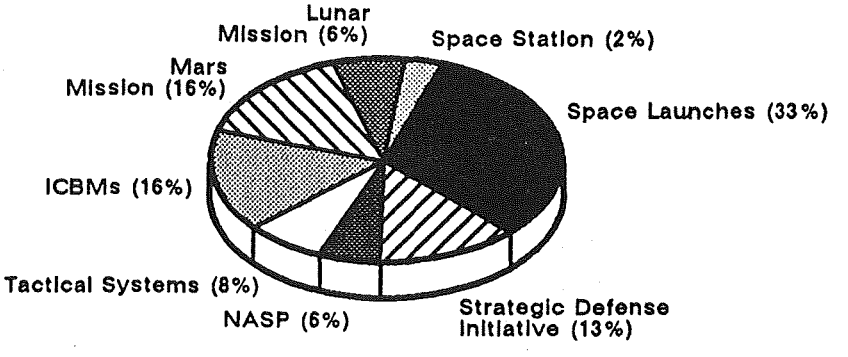


Figure 12. Projected Annual Costs



Each sector will benefit directly as the objectives of the *Strategic Plan* are met.

Figure 13. Benefit Distribution



Developed technologies will coincide with national goals.

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Figure 14. Benefits by Mission (including Department of Defense)

Such a national financial commitment cannot be short term. It must be renewed and sustained into the next century to meet future space and defense rocket propulsion needs (Figure 15).

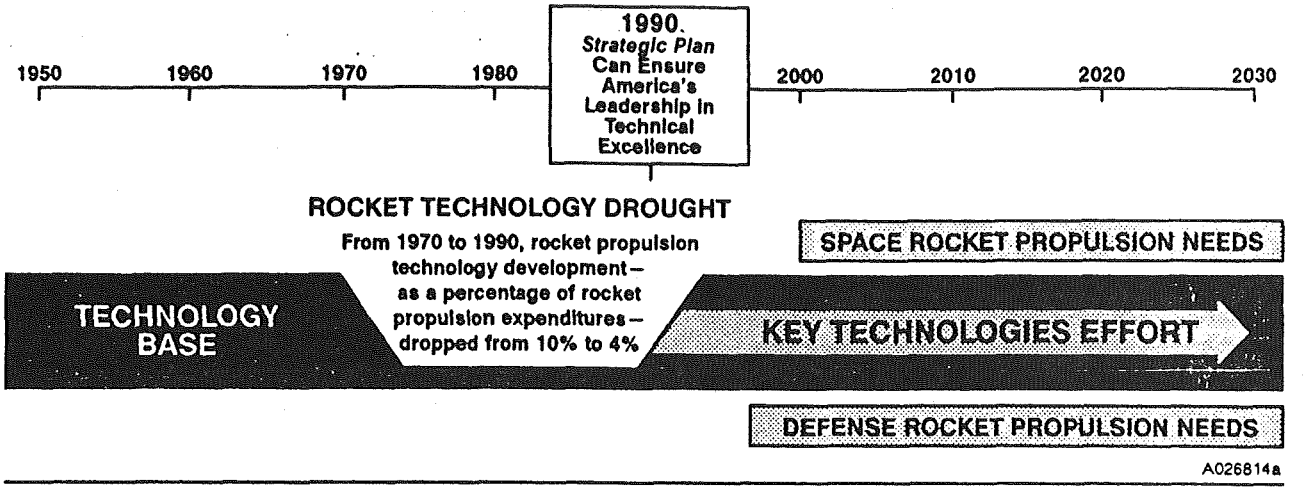


Figure 15. The Technology Drought

The *Strategic Plan* has a great deal of national leverage. When implemented it will power America into the future, as illustrated in Figure 16. To maintain technical excellence and global competitiveness, we must adopt the conclusion of the AIA Rocket Propulsion Committee and the planning team—PROPULSION TECHNOLOGY ISN'T EXPENSIVE: IT'S PRICELESS.

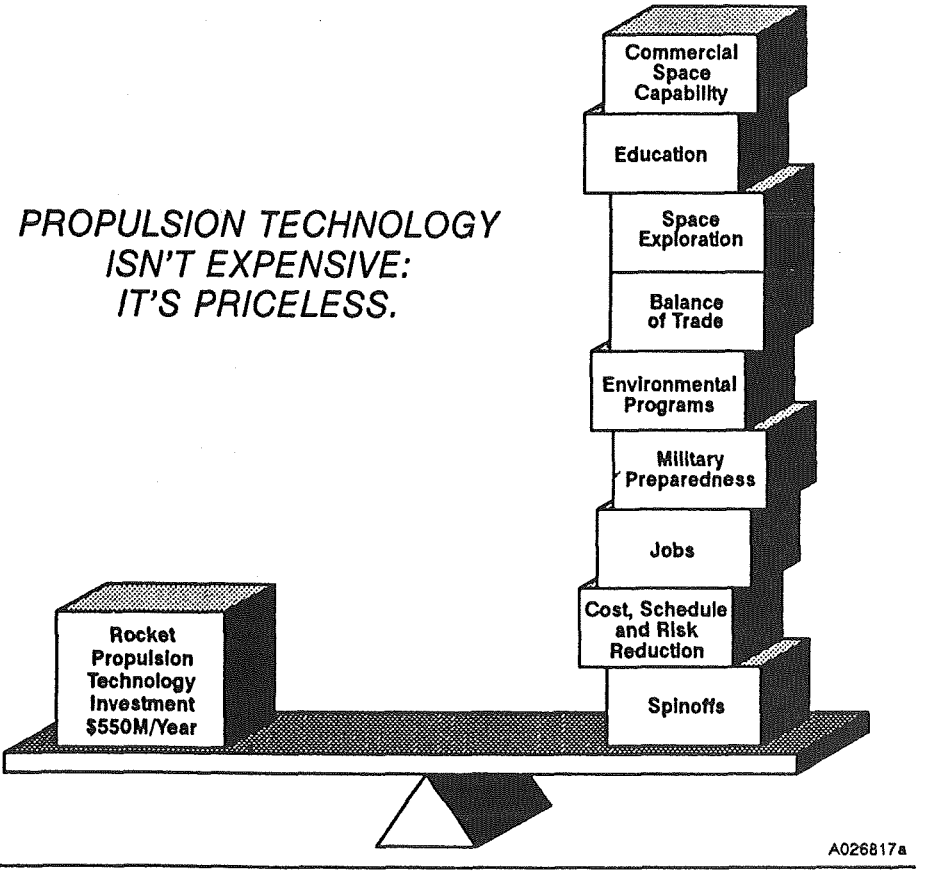


Figure 16. National Benefits

How can the plan be implemented? First, it will take unprecedented cooperation within the rocket community. Generating the plan has shown that it can be done. Second, the plan must be sold to decision makers in Congress, Government and industry. Starting last month and continuing through the summer, AIA has been briefing the *Strategic Plan* to Congressional and Government decision makers. Third, Government, industry and academia organizations must use the *Strategic Plan* as a basis, as applicable, for their own plans. Fourth, a mechanism must be established to coordinate industry, Government and university plans with the AIA *Strategic Plan*. A possible approach could be to use the JANNAF Executive Committee with industry participation.

Today rocket propulsion and technical excellence are at a crossroads. The comparison of our current position with that of the steel industry in the 1960s is frightening (Table 3). Rocket propulsion must not suffer the same fate as the steel industry.

Table 3. The Nation Is at a Rocket Propulsion Crossroads

	1960 Steel Industry	1990 Rocket Industry
Aging Work Force	X	X
Aging Facilities	X	X
Declining Technological Base	X	X

The implementation of the *Strategic Plan* will require:

- Rocket community cooperation
- Decision maker participation
- Inclusion in organization plans
- National coordination mechanism

Rocket propulsion must be a NATIONAL TECHNOLOGY IMPERATIVE. The time to act is now. The choice is decline or progress! For the first time, we now have a national rocket propulsion strategy. It needs your support and commitment. I am reminded of a quotation from C. J. Grayson's *Productivity, A New Scenario* that applies to rocket propulsion, technical excellence and global competitiveness:

> *"The crisis is real. For any leader, the time to worry is when your speed is slower than the horses coming up behind. The time to worry is not after but before they pass you by."*

OPERATIONAL EFFICIENCY - NEW APPROACHES TO FUTURE PROPULSION SYSTEMS

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A WHITE PAPER:

OPERATIONAL EFFICIENCY

NEW APPROACHES TO FUTURE PROPULSION SYSTEMS

RUSSEL RHODES AND GEORGE WONG
NASA Kennedy Space Center \ Rockwell International

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OPERATIONAL EFFICIENCY

NEW APPROACHES TO FUTURE PROPULSION SYSTEMS

FOREWORD

First, I would like to thank the Program Committee for giving the launch site the opportunity to provide visibility from our experience base back into the technology development process. I feel this is very important if we are to resolve these large deficiencies; they must be made visible.

Until now, our main thrust has been simply getting into and back from space. All criteria has been based on performance parameters, such as ISP, GLOW, T/W, mass fraction, etc. The rocket engine development, because of required long lead, led the process by establishing artificial interfaces for the design and operational control. The engine contract end item specification (CEI) and interface control document (ICD) were used for ease of procurement and development testing and to establish interface requirement for whoever desired its use. The vehicle, therefore, would assume the weight and operational burden of all the systems demanded by the engine. The mission use would determine the vehicle size and the number of engines required. Cost and launch rate were not of concern during the early years.

During the Apollo lunar exploration program, it became apparent that the Apollo vehicle launch operations were consuming a very large part of the agency budget, leaving very little for other scientific work and no new start programs. Therefore, we determined that developing a new vehicle that reused the very expensive vehicle hardware was the answer, ie., the Shuttle vehicle was born with expected large reductions in the cost of delivering a pound to orbit with 60 launches per year. Forty launches at KSC and 20 at WTR per year, but the design did not support this ambitious launch program. Also, the launch operations crew size was nearly the same as for the Apollo vehicle. Where did we fall short in our vision?

KSC initiated a self-examination three year study of cause and effect, led by Bill Dickinson and performed by the Boeing Company. This effort identified the vehicle configuration is the primary driver of this high cost limited launch capability. It also identified the propulsion system as a major discipline driver. Therefore, we initiated a more in-depth study of the causes and effects with the hope of identifying major generic operations concerns that cause the status quo. This present one-year effort has accomplished this, along with identifying alternate concepts that offer major reductions in complexity and manpower intensive operations. Therefore, the next 30 years we can focus on an ambitious space exploration by applying the knowledge gained from this visibility.

By applying the principles of TQM (old fashioned team effort) to Advance Planning, Conceptual Design, Development of Requirements and through the Design Development Process, we can achieve low cost, reliable, timely access to space and an operationally flexible space transfer system.

From our experience, the approach to follow is clear: Develop a simple, reliable, operationally efficient, integrated propulsion system concept that can be used and sized for different missions/vehicles. The concept must be fully integrated to achieve major reduction in propulsion components. This approach will yield major reduction in traditional vehicle support systems. We need to concentrate on the use of LOX/LH2 for all vehicle fluid needs. This combination will provide an environmentally clean operation and will enable a totally integrated propulsion and vehicle power capability, ie., MPS, OMS, RCS, fuel cells, cooling/thermal management and life support systems. Now, what is the propulsion development approach to follow?

First, we must surface the necessary technology needs to allow this ambitious space exploration program to occur. Develop these technology items into projects and follow them through maturity for use. I can't over stress the importance of a thorough maturation program, including flight tests in some cases. We must maximize the use of manpower and facilities. After all, the most valuable resource this country has is its people. We suggest we consider realigning our Government and industry teams and procurement practices to perform productive work and increase operational flexibility. We must discontinue our practice of creating artificial interfaces, unnecessary constraints, to allow fresh creative work to progress. After all, unnecessary constraints are the enemy of the bold. The competitive approach to advance planning and conceptual design is very wasteful; therefore, we suggest the consortium concept be considered. Let's use the competitive approach to providing high quality hardware from at least two sources.

Let us develop a means of measuring operability during our conceptual/design process. The commercial sector compares the use time to the shop maintenance/overhaul time and for them to turn a profit, this ratio must be in favor of use time. Traditionally, we spend large amounts of time preparing for a very short use time. Our conservative leadership is reluctant to make a long term commitment of advancing propulsion operations and give up their comfortable position of accepting the status quo, along with its near term personal or corporate gains. Can we afford to continue using the old patterns (ICD's and CEI's) while the rest of the world takes over the leadership position of space propulsion.

Let us accept the challenge for the future. Don't simply build a new model (an old one with a face lift) and spend 90% of our efforts concentrating on the lift off and ascent extravagance when it should be a routine event. But, instead, let us work together as a team and provide real measurable progress, allowing us to achieve the next frontier! "Routine Access to Space."

Mr. George Wong (Rocketdyne-Canoga Park, CA) will now talk to you about how applying the TQM team process makes a difference. He will share with you his experience this last year and give you an example of how this experience can influence the future of propulsion with focused technology development and the freedom to be creative.

Operationally Efficient Propulsion System

R.E. Rhodes and G.S. Wong

Introduction

Advanced launch systems for the next generation of space transportation systems (1995 to 2010) must deliver large payloads (125,000 to 500,000 lbs) to low earth orbit (LEO) at one tenth of today's cost, or 300 to 400 $/lb of payload. This cost represents an order of magnitude reduction from the Titan unmanned vehicle cost of delivering payload to orbit. To achieve this sizable reduction, the operations cost as well as the engine cost must both be lower than current engine systems. The Advanced Launch System (ALS) is studying advanced engine designs, such as the STME, which has achieved notable reduction in cost. This paper presents the results of a current study wherein another level of cost reduction can be achieved by designing the propulsion module utilizing these advanced engines for enhanced operations efficiency and reduced operations cost.

The operations cost of today's launch systems has become a large fraction of the vehicle recurring cost per flight ranging from 20 to 40% for expendable and reusable vehicles, respectively, shown in Figure 1. The complex operations requirements of current launch vehicles have also limited our ability to achieve routine access to space. Since the rocket engine/propulsion system represents one of the more complex and expensive systems in the launch vehicle, a study was made to identify operations problems (cause and effect concerns) which have driven operations costs to exorbitant levels. This paper presents the importance and a description of the major operations problems encountered in today's launch vehicles and how these problems have adversely affected our ability to achieve serviceability, reliability and operability. It also emphasizes the need to recognize and understand the operations problems and the effort that must be made to avoid them in future designs, i.e. applying the "lessons learned". It describes how the operations requirements for accessibility, maintainability and operability are allowed to start with the initial engine design to drive the design requirements. This has never been done before and this has been part of the reason today for the high cost vehicle launch systems and for the large launch processing cost and time. Finally, the paper presents an example whereby a propulsion concept that "integrates" the engine system not only results in a propulsion system that is more operationally efficient, with sizeable reduction in operations cost, but also results in a propulsion system that is simpler, more reliable, more operable and has lower cost than a conventional unintegrated engine system.

Current Operations Problems

Processing flight hardware for launch has been a very tedious and time consuming task requiring large numbers of people operating sophisticated ground support equipment (GSE) to verify flight system readiness. For each subsystem assembled with the major vehicle element, such as the Orbiter, comes the requirements for total system checkout prior to certification for flight. This process has been quite complex and involves numerous other systems during the checkout.

For Example, to support checkout of a main engine, the main propulsion system, electrical power and distribution system, hydraulic system, instrumentation system, flight control system, avionics system, environmental system and the purge, vent and drain

systems must all be activated to support the engine checkout. The checkout itself also requires highly trained and skilled personnel at the vehicle, in the firing room and at the GSE supplying the required commodities like gases, hydraulics, power, etc. All these activities are in turn dependent on test conductors, quality control, safety, GSE engineering, etc. to accomplish a successful test. As many of these activities are "hands-on" and serial in nature further complicates the checkout process. The ground support system providing services and commodities also must be verified that every system is available and certified to support the test. It is therefore not surprising that operations support for launch system checkout is complex, manpower intensive, time consuming and costly and a launch system that consists of many separate, independent systems simply exacerbates this problem.

A typical illustration of the technical disciplines and operations support required for system checkout is depicted in Figure 2. An illustration of the large infrastructure of logistics, supplies, equipment and facilities to support the system checkout is shown in Figure 3. Every different commodity required on the vehicle adds another tentacle to the operations support structure. For example, the requirement for Helium gas, no matter how small the amount, dictates the need for additional facilities, GSE, logistics, transportation, etc. to insure that the gas is at the vehicle processing site when needed.

Several recent studies on launch site experience have been made to identify operations problems that have driven our operations cost to exorbitant levels and have severely restricted our ability to achieve routine access to space. The Shuttle Ground Operations Efficiencies/Technologies Study (SGOE/T)[1] investigated the operations requirements of the entire vehicle including payload and the more recent "Operationally Efficient Propulsion System Study (OEPSS)[2] focused on the operations requirements of the total propulsion system that included: the propellant tankage, fluid systems, structure, engines and controls. Both studies have concluded that current operational requirements are driven by (1) systems that are not readily serviceable; (2) too many people are required; (3) too much time is needed for processing; (4) complex support facilities are needed; (5) serial operations are required; (6) hazardous operations are involved; (7) and too many commodities and grades of commodity are used.

The OEPSS study has also identified some serious major problems that have plagued our launch operations requirements and have compromised our launch capability. Figure 4 contains a list of these operations problems and the main propulsion system contained within a closed aft compartment was found to have the most widespread impact on ground operations. Other operations problems that drive operations support include the hydraulic systems, gimbal systems, turbopumps, inert gas purge, excessive number of components, many artificial interfaces and the lack of hardware integration. Some of these are described below.

Closed Aft Compartment

An enclosed engine compartment at the boat-tail of the launch vehicle causes numerous ground operations problems because leakage of hazardous fluids can be confined, access is restricted and complex GSE is required. Confinement of potential propellant leaks is a Criticality-1 failure. A closed compartment will require an inert gas purge system, a sophisticated hazardous gas detection system and a personnel environmental control system. These systems in turn will require vehicle - ground interfaces and ground support

equipment, all of which in turn will require separate specialized personnel to provide maintenance, checkout and servicing. Moreover, inert gas purge poses personnel safety issues.

Hydraulic System

A hydraulic system represents another fluid distribution system that must be processed and maintained for flight operations. This involves distribution system leak checks, long periods of circulation for de-aeration/filtering operations associated with fluid sampling and analysis, and functional check of all control systems. In order to process the flight system, a ground support system consisting of all the basic hydraulic distribution system elements must be duplicated to simulate pressure for the flight system checkout. The same operations and maintenance requirements are also required for the ground system.

The auxiliary power units to drive the hydraulic pumps represent an additional support system of prime mover, pumps, gearboxes, lube oil system, cooling system, instrumentation, distribution system, etc. which will require additional maintenance and checkout; and if a hypergolic-fueled auxiliary power unit is used, this will drive the need for a whole separate operations support infrastructure that dictates serial operations and the need for specially certified personnel to work in self-contained atmospheric protective ensemble (SCAPE) for fueling operations.

Lack of Hardware Integration

A launch system that contains numerous separate, stand-alone systems proportionally drives up the number of duplicate components and interfaces. This in turn exponentially drives up the complexity and the operational support requirements. Each stand-alone system promotes artificial interfaces and each interface represents another "break point" in the system that must be checked and verified should the connection be broken. Each fluid interface represents a potential leak point requiring special attention for disassembly, reassembly and leak checks. Separating fluid connections leads to potential sealing surface damage, which in turn requires repair of the sealing surface and, if severe, requires a line changeout. It is not uncommon in a critical system containing helium, hydrogen or oxygen to replace seals more than once to ensure an acceptable leak-free joint. An example of separate stand-alone systems is a launch vehicle propulsion system using multiple autonomous engines. The propulsion system will have as many duplicate propellant lines, valves, thrust chambers, turbopumps, control/avionics, heat exchangers, pneumatic control assembly, etc and interfaces as there are engines.

Systems carrying fluids such as hydrogen and oxygen necessarily dictate the use of sophisticated, highly sensitive, operations intensive leak detection devices, such as mass spectrometers, to verify the integrity of the seal. This requirement drives up the time required to leak check a joint considerably. High helium content in the surrounding area can cause leak checks to be delayed until the background is reduced or add time to the operation by having to encapsulate each joint that is checked. Leak checking many joints has led to time-consuming serial operations impacting the total system checkout.

In view of current experience, it is abundantly clear that operational complexity stems from design. The operational support of current flight systems was never fully understood nor the impact on launch processing was fully appreciated during design. In order to achieve operational efficiency, the principle of Total Quality Management (TQM) must be

applied to ground operations as it is being applied to product quality, that is quality cannot be inspected into the product, it must be designed into it. Therefore, operations must not simply support the design it must change and drive the design at its conceptual beginning toward greater simplicity and greater operability. This imperative approach is illustrated in the design/build/operations cycle shown in Figure 5.

Operationally Efficient Propulsion System

To achieve operational efficiency for a flight system the design must be simplified to reduce operations required to support the system. An example will be used here to illustrate how the "lessons learned" from current operations experience (Figure 4) are used to drive the design of a propulsion system concept for a heavy lift launch vehicle, such as the Advanced Launch System (ALS). The example will describe how the design can be simplified by "integrating" the multiple engines to eliminate as many components and interfaces as possible while maintaining the required thrust and control of the vehicle.

The baseline ALS vehicle shown in Figure 6 will be used as a reference vehicle for comparing a traditional approach to designing a conventional propulsion system vis-a-vis with an integrated approach to designing an operationally efficient propulsion system. The ALS vehicle shown consist of a core vehicle and a side-mounted booster with a gross lift-off weight (GLOW) of 3,500,000 lbs. and a payload capability of 120,000 lbs. to low earth orbit (LEO). Both the booster and core vehicles are 30 ft. in diameter and use 580,000 lbs. thrust (vac) O_2/H_2 STME engines. The booster and core utilize 7-engines and 3-engines, respectively, for their propulsion systems.

Summary and Conclusion

Today's launch systems have resulted in high operations cost and low flight rates. The complex systems have been found to be the cause for the inordinate time and manpower needed to meet ground operations requirements and for our inability to achieve routine access to space. The complex propulsion system for our current launch systems has been a major part of this problem. In order for future advanced launch vehicles, such as the ALS, to deliver payload to orbit (LEO) at lower cost and at higher flight rates, the design of the launch system, and particularly the propulsion system, must be greatly simplified and made more operationally efficient. The results of the current study ~~summarized in Figure 18~~ have shown that by utilizing an unconventional "integrated" design approach, a low cost, operationally efficient propulsion system design can be achieved. Based on the study results, the following conclusions are made:

(1) To achieve an operationally efficient, low cost propulsion design, operations cost drivers must drive the design at the inception of concept. A design that initially ignores operations problems can not subsequently be made truly operationally efficient.

(2) Propulsion system design for future launch systems can be made simpler and require less operations support by reducing the number of components and interfaces and by integrating the system functions. This is achieved by departing from the conventional engine design approach and by using the "integrated-component" design approach described.

(3) The integrated propulsion module engine as an alternative propulsion concept for the ALS illustrates the following point: given a propulsion system design using multiple stand-alone, autonomous engines, an integrated design of the same system will always yield an equivalent system that will have substantially higher reliability and lower unit cost.

(4) An integrated propulsion design is tractable[3] and can use existing or current ALS technology and does not require new technology (enabling).

(5) An integrated design approach results in a propulsion design that is simpler, more reliable, more operable, lower unit cost than a conventional design and, therefore, eminently meets the ALS requirements for robusmess, reliability, operability, low cost and the ability to achieve routine access to space.

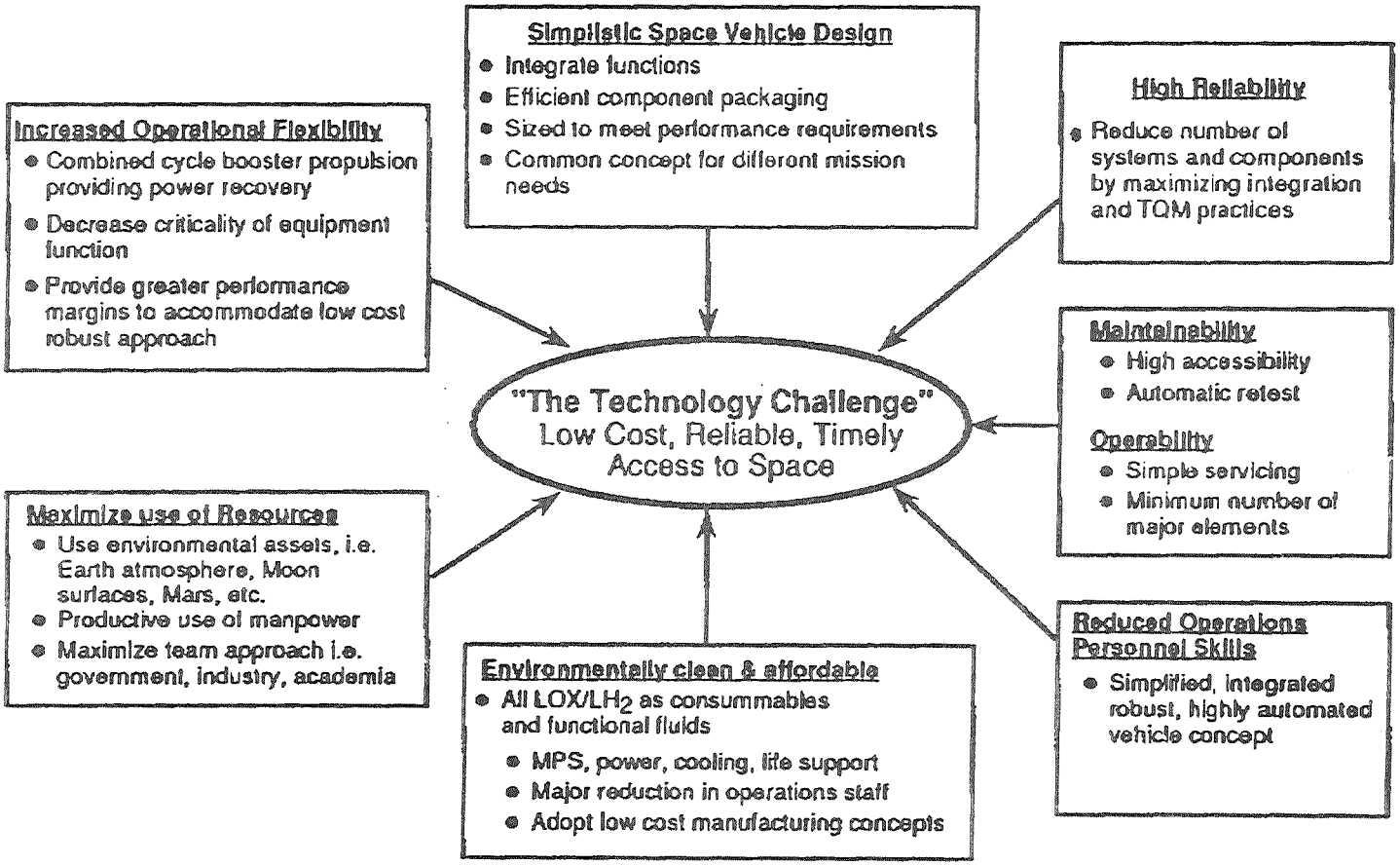
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2. "Operationally Efficient Propulsion System Study" (OEPSS), NASA/KSC Contract NAS10-11568, G.S. Wong, Rocketdyne Division, Rockwell International, RI/RD90-149 (to be issued), 1990

3. "A New Look at Chemical Rocket Propulsion System Configurations for Space - Stage Transport Systems", W.J.D. Escher, Propulsion, Power and Energy Division, NASA Headquarters (informal discussion paper) March 1990

FUTURE PROPULSION DEVELOPMENT



PROPULSION SYSTEM FOR ALS

- **Defined as a totally "integrated" system of components and subsystems to provide vehicle thrust and control**
 - Tankage
 - Fluid Systems
 - Structure
 - Thrust Chamber(s)
 - Turbopump(s)
 - Controls
- **Use a "minimum" of components and subsystems to meet the functions of the propulsion system**
 - Simple
 - Reliable
 - Robust
 - Operationally efficient
- **Achieve lowest possible cost by applying TQM to propulsion system development process**
 - Design/Build/Operate

Figure 1

LAUNCH VEHICLE OPERATIONS COST PER FLIGHT

% of Total Recurring Cost

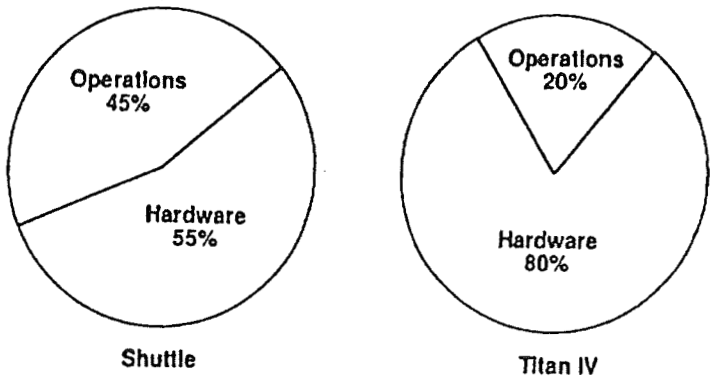


Figure 2

OPERATIONS SUPPORT

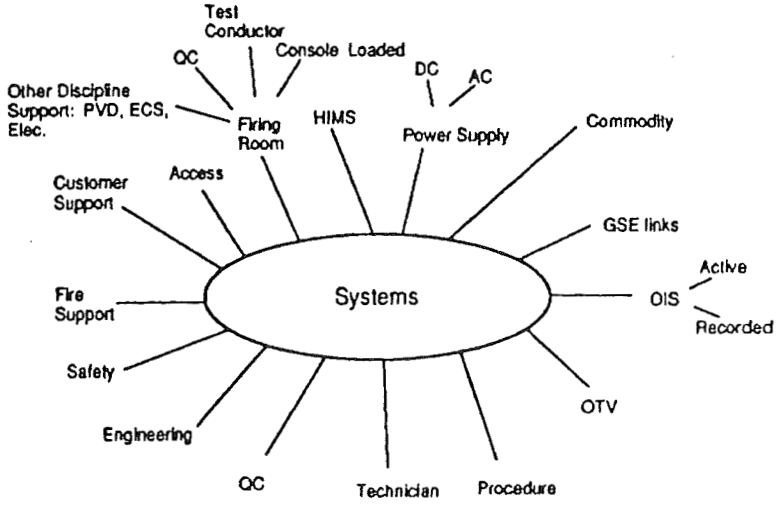


Figure 3

OPERATIONS SUPPORT STRUCTURE

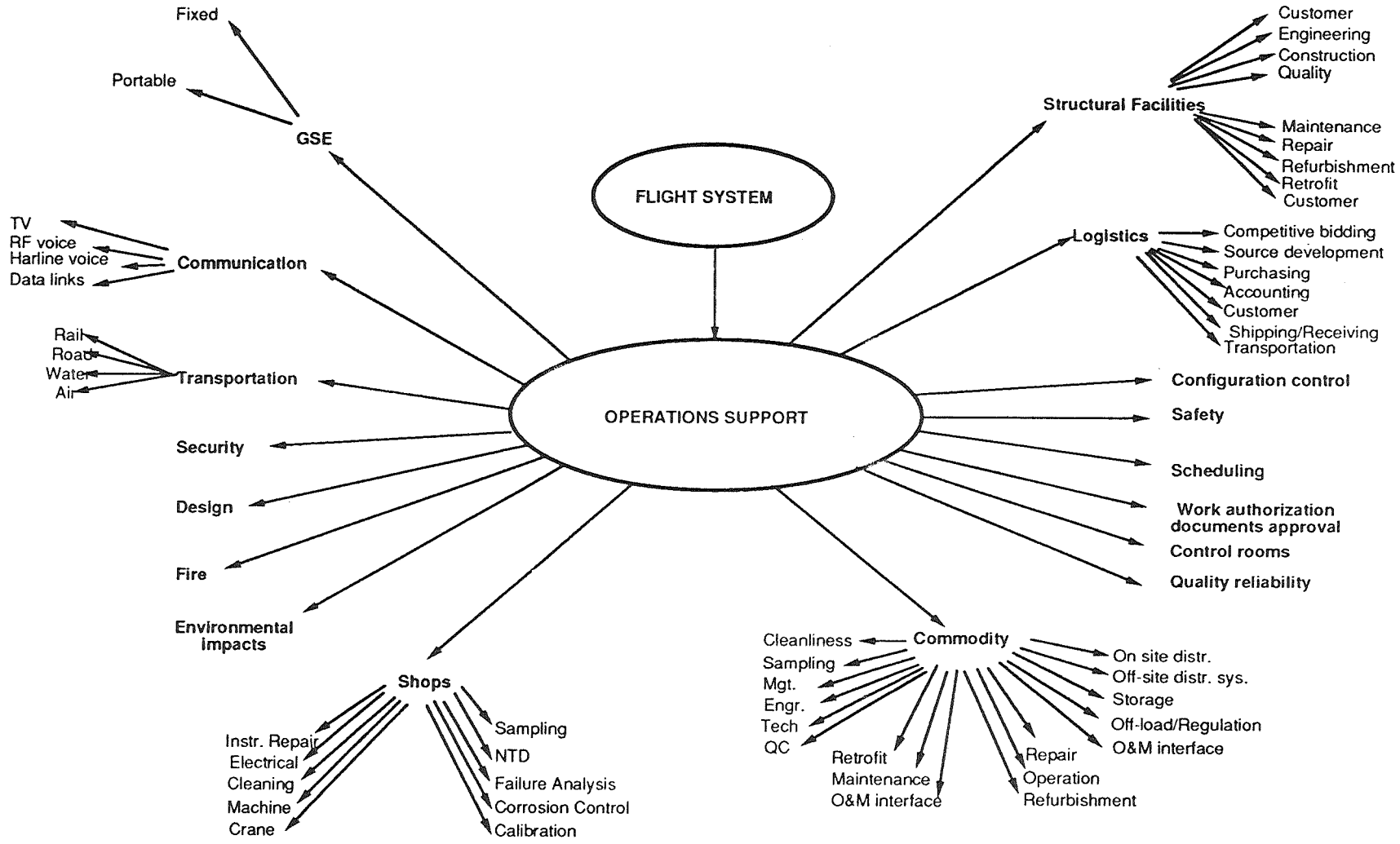


Figure 4

OPERATIONS PROBLEMS CAUSE AND EFFECTS

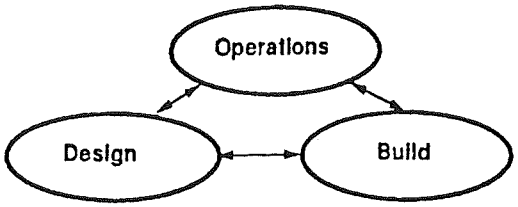
No.	
1	Closed aft compartments
2	Hydraulic system for valve actuators and TVC
3	Ocean recovery and refurbishment
4	Multiple propellants
5	Hypergolic propellant safety
6	Accessibility
7	Sophisticated heat shielding
8	Excessive components/subsystem interfaces
9	Lack of hardware integration
10	Separate OMS and RCS
11	Pneumatic system for valve actuators
12	Gimbal system requirements
13	High maintenance turbopumps - recoverable propulsion system

No.	
14	Ordnance Operations
15	Retractable T-O umbilical carrier plates
16	Pressurization systems
17	Inert gas purging requirements
18	Numerous interfaces
19	Helium spin start
20	Liquid oxygen tank forward design (propellant system geometry)
21	Preconditioning system
22	Expensive commodity usage - helium
23	Lack of hardware commonality
24	Contamination
25	Side-mounted booster launch vehicles (multiple stage element propulsion systems)

Figure 5

TOTAL QUALITY MANAGEMENT (TQM)

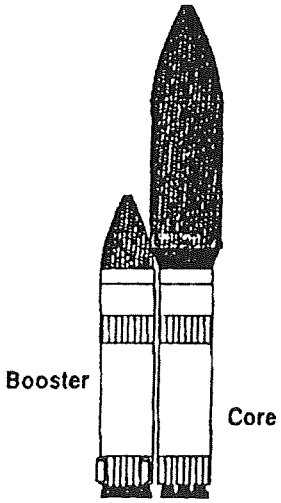
For Total Propulsion System



OEPSS

Figure 6

BASELINE ALS VEHICLE



• Payload	120,000 lbs (LEO)
• GLOW	3,500,000 lbs
• Thrust/weight	1.30
• Booster vehicle	150' x 30' dia.
• Core vehicle	280' x 30' dia.
• Booster engines	7
• Core engines	3
• Engine thrust (vac)	580,000 lbs (STME)

Figure 7

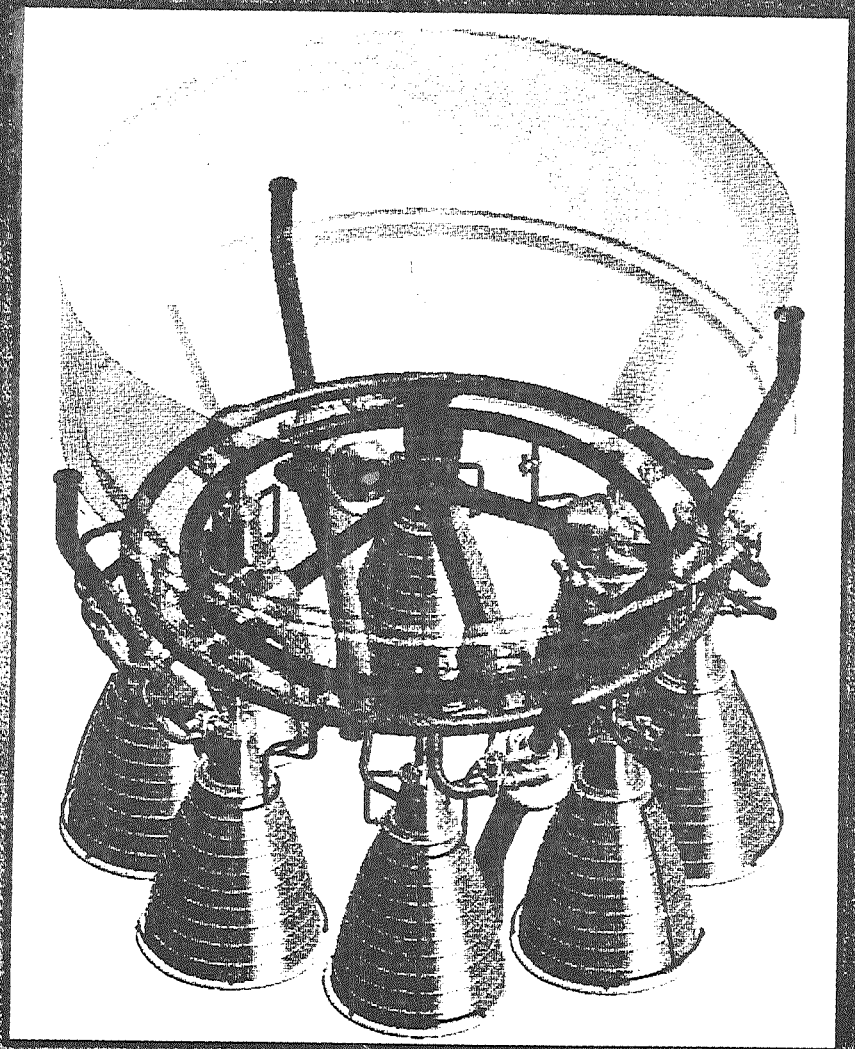
CONVENTIONAL BOOSTER PROPULSION SYSTEM
7 - ENGINE

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Figure 8

INTEGRATED BOOSTER PROPULSION MODULE – ENGINE

8 – THRUST CHAMBERS



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Rocketdyne Division

SE-89-30-153A

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Figure 9

FULLY INTEGRATED PROPULSION MODULE

- Single He-pressurization system*
- Single LOX-pressurization system* (HX)
- Single control system*
- Torus propellant manifold allows 50% reduction of
 - Turbopumps
 - Propellant inlet lines
 - Gas generators
- Torus manifold provides "engine-out" capability
 - Thrust chamber-out
 - Turbopump-out

*Redundancy provided in propulsion module

Figure 10

"ROBUST ENGINE AND ENGINE OUT" CAPABILITY

- Thrust chamber out capability
 - Thrust chamber 85% ——→ 100% Nom. Oper.
 - Turbopumps 67%
- Turbopump out capability
 - Turbopumps 67% ——→ 100% Nom. Oper.
 - Thrust chamber 85%

Figure 11

ROBUST TURBOPUMP DESIGN

- Design margin
- Operating margin

Booster	7-engine (7-T/P)	8-engine (4-T/P)	
	Des. RPM (100%)	Des. RPM (100%)	Oper. RPM (67%)
LH2-Turbopump	26,000	18,600	12,500
LO2-Turbopump	10,000	7,100	4,800

Figure 12

SUMMARY OF RESULTS

- Integrated propulsion module vs. conventional propulsion system

Factor	Fully Integrated	Conventional
• Higher reliability	0.993	0.987
T/C and T/P out	0.999	-----
• Lower engine (T/C) cost, $M	1.83	2.67
• Less number of parts	111	169
• Lower potential weight, lbs.	76,058	87,340
• Lower operations cost, %	-35 to -60	-----

Figure 13

OPERATIONS CONCERNS RESOLVED BY TECHNOLOGY

OEPSS Concerns: ① ② ③ ④ ⑤ ⑥ ⑦ ⑧ ⑨ ⑩ ⑪ ⑫ ⑬ ⑭ ⑮ ⑯ ⑰ ⑱ ⑲ ⑳ ㉑ ㉒ ㉓ ㉔ ㉕

Technology	OEPSS Concerns Addressed
• No purge pump seals	⑧ ⑰ ⑱ ㉒
• No purge combustion chamber (start-shutdown)	⑧ ⑰ ⑱ ㉑ ㉒
• Oxidizer-rich turbine, LOX turbopump	⑧ ⑬ ⑰ ㉒
• Hermetically sealed inert engine and tanks (prelaunch)	⑧ ⑰ ⑱ ㉑ ㉒ ㉔
• Combined O_2/H_2, MPS, OMS, RCS, fuel cell, thermal control systems	④ ⑤ ⑧ ⑨ ⑩ ㉔
• Flash boiling tank pressurization	⑧ ⑨ ⑯ ⑱ ㉔
• Low NPSH pumps	⑬ ⑯ ⑲ ㉑
• Large flow range pumps	⑧ ⑬ ⑲ ㉑
• Differential throttling	② ⑦ ⑧ ⑫
• Electric Motor Actuator (EMA)	② ⑥ ⑨ ⑪ ⑫ ⑮ ⑱ ㉔
• No leakage mechanical joints	① ② ③ ⑤ ⑧ ⑪ ⑯ ⑰ ⑱ ㉑
• Automated, self-diagnostic, condition monitoring system	① ⑥ ⑪ ⑫ ⑬ ⑮ ㉑ ㉔
• Integrated modularized propulsion module concept	① ② ③ ⑥ ⑦ ⑧ ⑨ ⑪ ⑫ ⑬ ⑱ ⑲ ㉑ ㉒ ㉓ ㉔
• Anti-geyser, LOX tank aft propulsion concept	⑨ ⑲ ⑳ ㉑
• Rocket engine, air augmented afterburning concept	③ ⑨ ⑯ ⑰

Figure 14

OPERATIONS TECHNOLOGY APPLICATION

Technology	Vehicle Systems						
	STS	Sh-C	LRB	ELV	ALS	Sh-II	Space
• No purge pump seals			X		X	X	X
• No purge combustion chamber (start-shutdown)			X	X	X		X
• Oxidizer-rich turbine, LOX turbopump			X		X	X	X
• Hermetically sealed inert engine and tanks (prelaunch)			X		X		
• Combined O_2/H_2 MPS, OMS, RCS, fuel cell, thermal control systems		X			X	X	X
• Flash boiling tank pressurization			X		X	X	X
• Zero - NPSH pumps			X	X	X	X	X
• Large flow range pumps			X		X	X	X
• Differential throttling					X		
• Electric Motor Actuator (EMA)	X	X	X	X	X	X	X
• No leakage mechanical joints			X	X	X		X
• Automated self-diagnostic condition monitoring system	X	X	X	X	X	X	X
• Integrated modularized propulsion module concept		X	X		X	X	X
• Anti-geyser, LOX tank aft propulsion concept			X		X	X	
• Rocket engine, air augmented afterburning concept			X		X	X	

Figure 15

CONCLUSION

- **Operations efficiency requirements must start with the initial system design**
- **Operations efficiency to reduce cost must drive the system design in a TQM team environment**
 - Design / build / operate
- **The integrated propulsion module engine is only one example where:**
 - The opportunities for higher operational efficiencies were more fully explored
 - The measurable gains in operational efficiency were identified
- **Other propulsion concepts exist for which the possibilities of greater operational efficiencies have not been fully explored**

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SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM

OPERATIONALLY EFFICIENT PROPULSION SYSTEM STUDY (OEPPS)

The Pennsylvania State University
University Park, PA
25 - 29 June 1990

OPERATIONALLY EFFICIENT PROPULSION SYSTEM STUDY (OEPSS)

NAS 10-11568

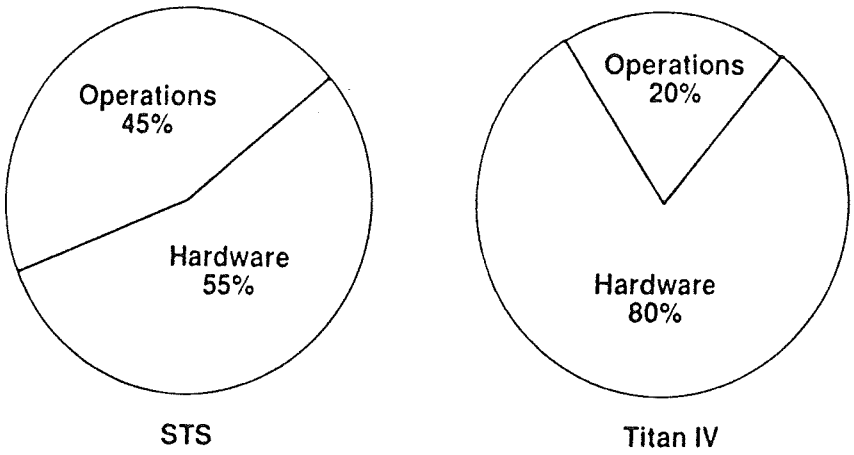
April 1989 - April 1990

G.S. Wong, G.S. Waldrop, R.J. Byrd, J.M. Ziese

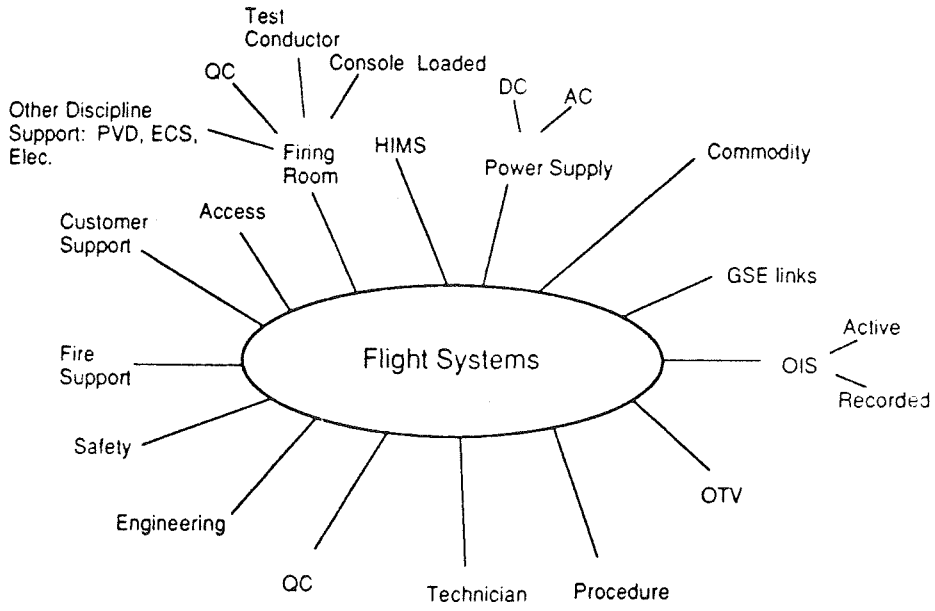
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LAUNCH OPERATIONS COST PER FLIGHT

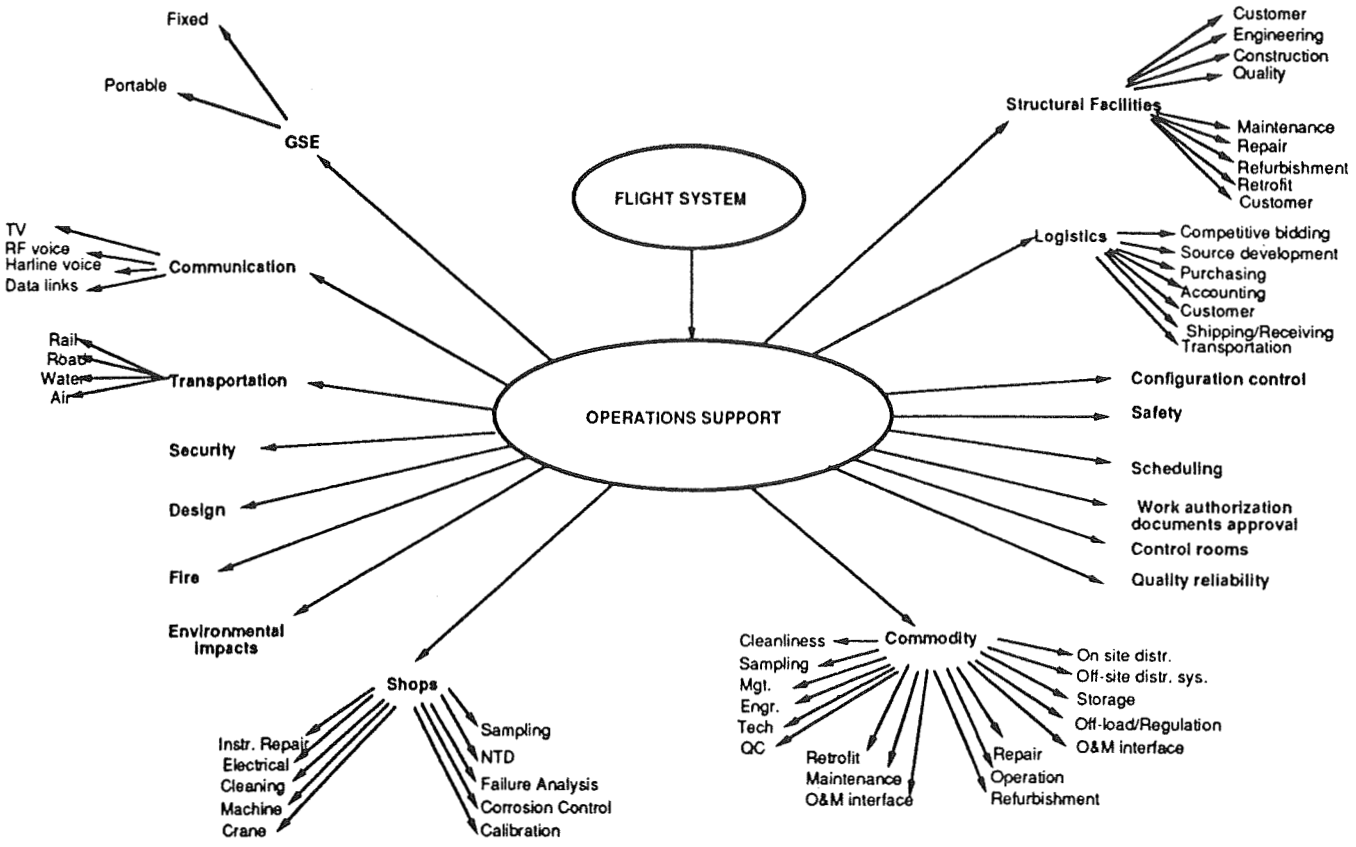
% of Total Recurring Cost



OPERATIONS SUPPORT SYSTEM IS COMPLEX



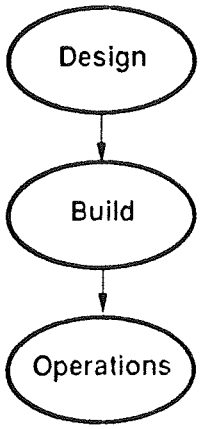
OPERATIONS SUPPORT STRUCTURE IS COMPLEX



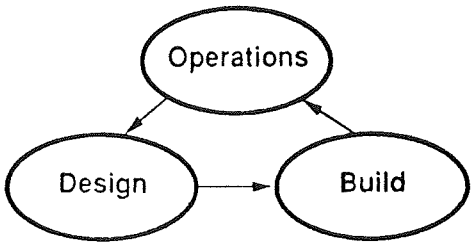
OPERATIONS PROBLEMS RESULTS IN HIGH COST

- Operations problems largely ignored
- Operations is a major cost driver
- Operations must play interactive role with propulsion system design

OPERATIONS AND DESIGN MUST BE INTERACTIVE



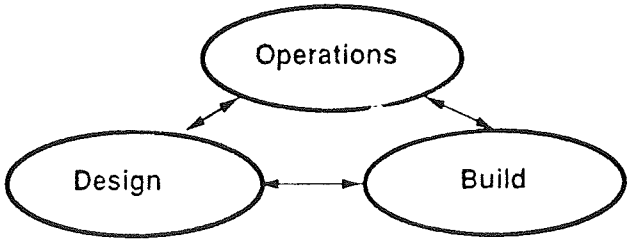
Traditional



OEPSS

TOTAL QUALITY MANAGEMENT (TQM) FOR OPERATIONS

Total Propulsion System



OEPSS

OEPSS IDENTIFIES OPERATIONS PROBLEMS

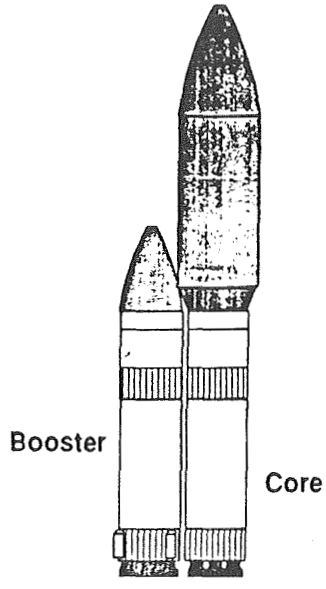
Causes and Effects

No.		No.	
1	Closed aft compartments	14	Ordnance Operations
2	Hydraulic system (valve actuators and TVC)	15	Retractable T-O umbilical carrier plates
3	Ocean recovery/refurbishment	16	Pressurization system
4	Multiple propellants	17	Inert gas purge
5	Hypergolic propellants (safety)	18	Excessive interfaces
6	Accessibility	19	Helium spin start
7	Sophisticated heat shielding	20	Conditioning geysering (LO_2 tank forward)
8	Excessive components/subsystems	21	Preconditioning system
9	Lack hardware integration	22	Expensive helium usage - helium
10	Separate OMS/RCS	23	Lack hardware commonality
11	Pneumatic system (valve actuators)	24	Propellant contamination
12	Gimbal system	25	Side-mounted booster vehicles (multiple stage propulsion systems)
13	High maintenance turbopumps		

CURRENT OPERATIONS IS SERIAL, TIME CONSUMING AND MANPOWER INTENSIVE

- **Some major operations problems**
 - Closed boat-tail compartment
 - Hydraulic and gimbaling systems
 - Multiple propellants/commodities (LO_2, LH_2, hypergols, He, N_2, freon, etc)
 - Excessive components and interfaces
- **Reduce operations problems by integrating engine components and subsystems**
 - Integrated propellant feed and engine system
 - Integrated engine supports systems
 - Helium
 - Pressurization
 - Control avionics
 - Common O_2/H_2 systems
 - MPS
 - OMS/RCS
 - Fuel cells
 - ECLSS

BASELINE ALS VEHICLE



- Payload 120,000 lbs (LEO)
- GLOW 3,500,000 lbs
- Thrust/weight 1.30
- Booster vehicle 150' x 30' dia.
- Core vehicle 280' x 30' dia.
- Booster engines 7
- Core engines 3
- Engine thrust (vac) 580,000 lbs (STME)

Rockwell International

CONVENTIONAL BOOSTER PROPULSION SYSTEM

F-1 ENGINE

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CONVENTIONAL CORE PROPULSION SYSTEM
3 - ENGINE

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ALS INTEGRATED BOOSTER PROPULSION MODULE

- **Moved center engine to perimeter**
 - Eliminate potential pogo problem
 - Achieves accessibility and commonality
- **Eliminated components and interfaces**
- **Integrated He supply system**
- **Integrated pressurization system**
- **Integrated control/avionics**

Rockwell International

INTEGRATED BOOSTER PROPULSION MODULE – ENGINE

3 – THRUST CHAMBERS

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INTEGRATED CORE PROPULSION MODULE – ENGINE

4 – THRUST CHAMBERS

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INTEGRATED PROPULSION MODULE – ENGINE ELEMENT

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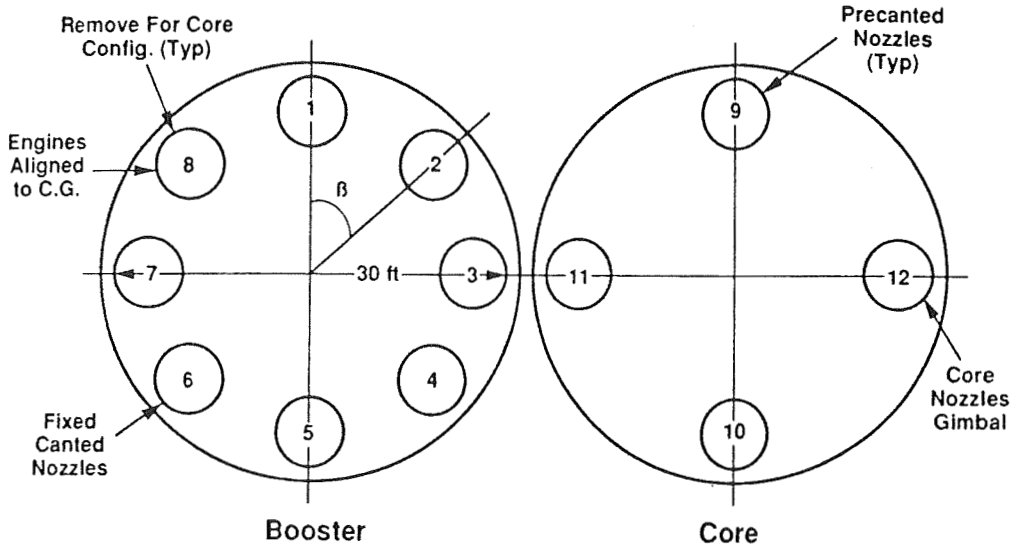
INTEGRATED PROPULSION MODULE DESIGN INCREASES OPERATIONS EFFICIENCY

- Single He-pressurization system
- Single LOX pressurization system (heat exchanger)
- Single control system
- No flexible propellant lines
- No gimbal actuators
- Torus propellant manifold allows 50% reduction of
 - Propellant inlet lines
 - Turbopumps
 - Gas generators
- Torus manifold provides "engine-out" capability
 - Thrust chamber-out
 - Turbopump-out

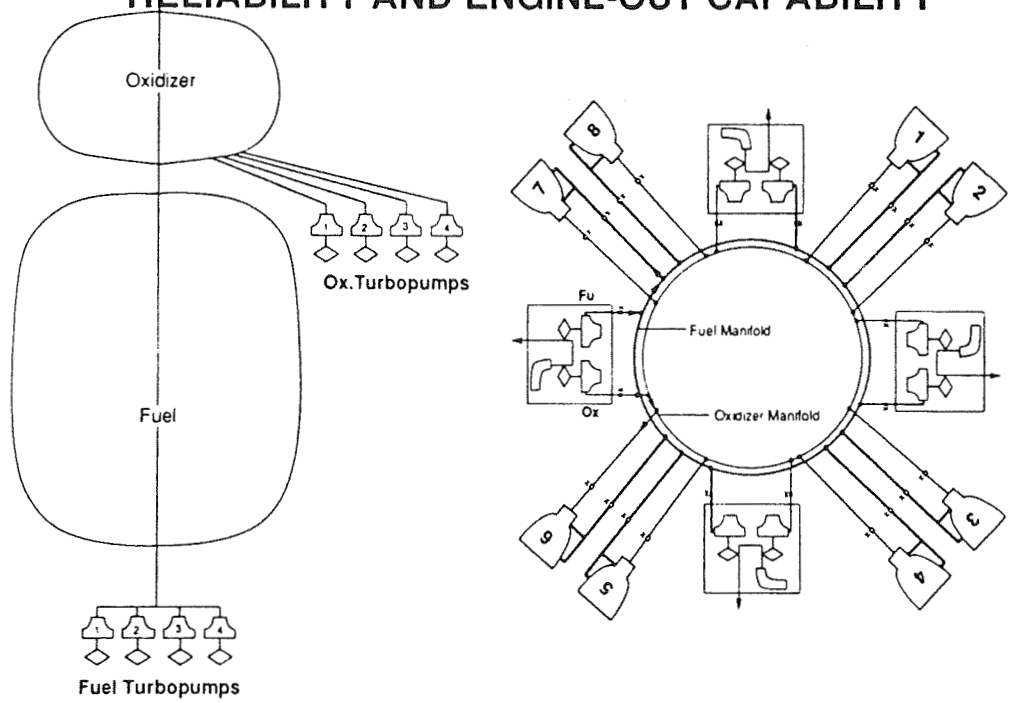
INTEGRATED PROPULSION MODULE MAXIMIZES ROBUSTNESS AND COMMONALITY

- Booster utilizes non-gimbaling thrust chambers: 8 T/C's
- Core provides TVC with gimbaled thrust chambers: 4 T/C's
- Normal engine operation at 85% nominal thrust
- Engine operates at 100% thrust with "engine-out" (1-T/C, 1-T/P)
- Outer thrust chamber arrangement maximizes maintainability
- Booster-core configuration achieves maximum commonality
 - Identical module thrust structure
 - Identical feedlines and valves
 - Identical thrust chambers
 - Identical turbopumps

8/4 BOOSTER-CORE CONFIGURATION ACHIEVES MAXIMUM COMMONALITY



INTEGRATED CONCEPT INCREASES RELIABILITY AND ENGINE-OUT CAPABILITY



INTEGRATED PROPULSION MODULE "COMPONENT OUT" CAPABILITY

- **Thrust chamber-out capability**
 - Thrust chamber 85% ⟶ 100% Nom. Oper.
 - Turbopumps 90% ⟶ 97% Nom. Oper.
- **Turbopump out-capability**
 - Turbopumps 90% ⟶ 93% Nom. Oper.
 - Thrust chamber 85%

INTEGRATED PROPULSION MODULE COMPONENT-0UT CAPABILITY

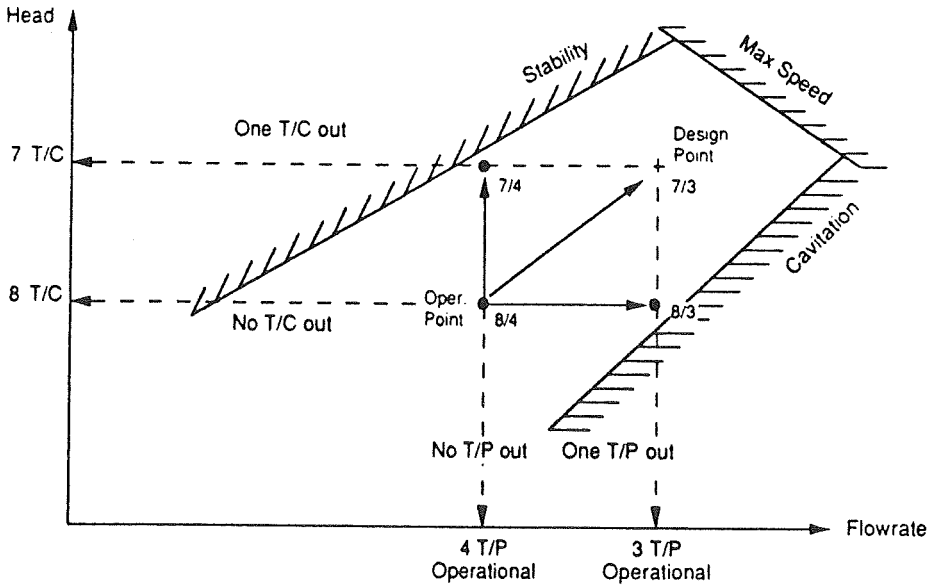
Engine Operation	Thrust Chamber (T/C) % Rated Thrust	Turbopumps (T/P) % Rated Speed
Nominal	85	90
T/C - Out	100	97
T/P - Out	85	93
T/C and T/P-Out	100	100

ROBUST TURBOPUMP DESIGN

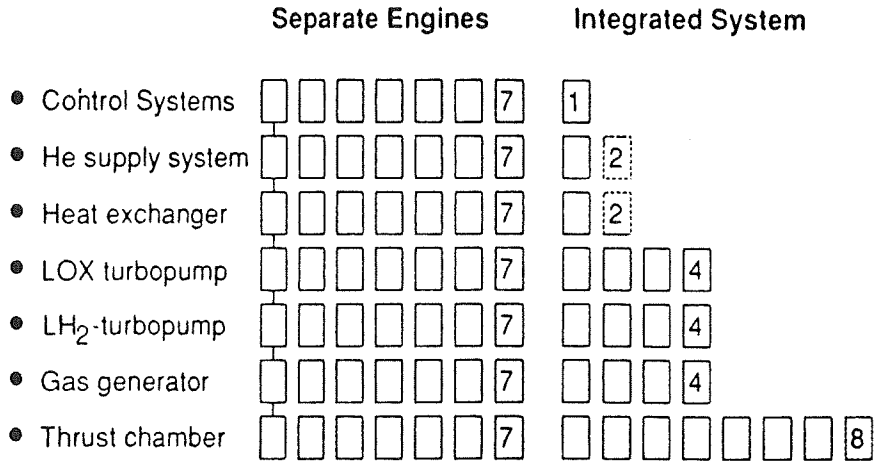
- Lower design speed
- Operating margin

Booster	7-engine (7-T/P)	8-thrust chamber (4-T/P)	
	Des. RPM (100%)	Des. RPM (100%)	Oper. RPM (90%)
LH2-Turbopump	26,000	18,600	16,700
LO2-Turbopump	10,000	7,100	6,400

TURBOPUMP OPERATING MAP
Integrated Propulsion System



SEPARATE ENGINES VS. INTEGRATED SYSTEM



BOOSTER PROPULSION MODULE HARDWARE COMPARISON

Separate Engines vs. Integrated System

Engine Elements	Separate Engines	Integrated System (Static)
	No. of Components	No. of Components
Thrust chamber:		
MCC	7	8
Injector	7	8
Nozzle	7	8
Igniter	7	8
Oxidizer turbopump	7	4
Fuel turbopump	7	4
Gas generator	7	4
Heat Exchanger	7	2
Start System	7	1
PCA	7	1
Controller (avionics)	7	1
Gimbal bearing	7	0
Gimbal actuator	14	0
Propellant lines	14	4
Flexible inlet lines	14	0
Fixed inlet lines	0	8
Main valve/actuator	14	24
Prevalves	14	0
Crossover duct/lines	7	0
HP T/P discharge lines	0	8
Ring manifold	0	2
HP T/C inlet lines	0	8
Miscellaneous	7	8
Center engine mount	1	0
Total	169	111

Table 1

BOOSTER PROPULSION MODULE RELIABILITY

Separate Engines vs. Integrated System

Engine Elements*	Component Reliability	Separate Engines		Integrated system	
		No. of Components	Subsystem Reliability	No. of Components	Subsystem Reliability
Thrust chamber assy	0.99978	7	0.99846	8	0.99824
T/C, ISO valve, ox	0.99996	0	-	8	0.99968
T/C ISO valve, fuel	0.99996	0	-	8	0.99968
Oxidizer turbopump	0.99986	7	0.99902	4	0.99944
Fuel turbopump	0.99972	7	0.99804	4	0.99888
MOV	0.99996	7	0.99972	4	0.99984
MFV	0.99996	7	0.99972	4	0.99984
Gas generator	0.99983	7	0.99881	4	0.99932
PCA	0.99999	7	0.99993	1	0.99999
Controller	0.99996	7	0.99972	1	0.99996
Gimbal system	0.99999	7	0.99993	0	-
Heat exchanger	0.99989	7	0.99923	2	0.99978
Propellant lines	0.99999	14	0.99986	4	0.99996
Inlet line, flex	0.99980	7	0.99860	0	-
Inlet line, fixed	0.99980	7	0.99860	4	0.99920
Prevalve, oxid	0.99996	7	0.99972	0	-
Prevalve, fuel	0.99996	7	0.99972	0	-
Crossover duct	0.99980	7	0.99860	0	-
HP T/P discharge lines	0.99999	0	--	8	0.99992
Ring manifold	0.99991	0	-	2	0.99982
HP T/C inlet lines	0.99999	0	-	8	0.99992
Overall reliability		0.98775		0.99351	

*STME Components

Table 2

BOOSTER PROPULSION MODULE SYSTEM WEIGHT

Separate Engines vs. Integrated System

Engine Elements	Unit Weight Lbs	Separate Engines		Integrated System	
		No. of Components	Weight Lbs	No. of Components	Weight Lbs
Thrust chamber:					
MCC	613	7	4291	8	4904
Injector	364	7	2548	8	2912
Nozzle	2088	7	14616	8	16704
Igniter	31	7	217	8	248
Oxidizer turbopump	1726	7	12082	4	9664 (1)
Fuel turbopump	1421	7	9947	4	7960 (1)
Gas generator	121	7	847	4	484 (2)
Heat Exchanger	101	7	707	2	404 (3)
Start System	35	7	245	1	70 (3)
PCA	82	7	574	1	82
Controller (avionics)	20	7	140	1	20
Gimbal bearing	158	7	1106	0	0
Gimbal actuator	190	14	2660	0	0
Propellant lines	--	14 (1186)	16600	4 (1587)	6348
Flexible inlet lines	734	14	10276	0	0
Fixed inlet lines	668	0	0	8	5344
Main valve/actuator	144	14	2016	24	3456
Prevalve	75	14	1050	0	0
Crossover duct/lines	214	7	1498	0	0
HP T/P discharge lines	360	0	0	8	2880
Ring manifold	3750	0	0	2	7500
HP T/C inlet lines	300	0	0	8	2400
Miscellaneous	585	7	4095	8	4680
Center engine mount	1826	1	1826	0	0
Total Weight			**87,340**		**76,058**

(1) Factor of 1.4. (2) Factor of 1.5. (3) Factor of 2.0

Table 3

INTEGRATED PROPULSION MODULE IS RELIABLE AND LOW COST

Factor	Separate	Integrated
• Higher reliability	0.988*	0.993*
T/C and T/P out	0**	0.999**
• Lower engine (T/C) cost, $M	2.67	1.83
• Less number of parts	169	111
• Lower potential weight, lbs.	87,340	76,058
• Lower operations cost	1	1/3

* No engine-out capability

** With T/C and T/P - out capability

INTEGRATED DESIGN ADDRESSES OPERATIONS PROBLEMS DIRECTLY

No.	
①	Closed aft compartments
②	Hydraulic system (valve actuators and TVC)
③	Ocean recovery/refurbishment
4	Multiple propellants
5	Hypergolic propellants (safety)
⑥	Accessibility
⑦	Sophisticated heat shielding
⑧	Excessive components/subsystems
⑨	Lack hardware integration
10	Separate OMS/RCS
⑪	Pneumatic system (valve actuators)
⑫	Gimbal system
⑬	High maintenance turbopumps

No.	
14	Ordnance Operations
15	Retractable T-O umbilical carrier plates
16	Pressurization system
17	Inert gas purge
⑱	Excessive interfaces
19	Helium spin start
20	Conditioning/geysering (LO_2 tank forward)
㉑	Preconditioning system
㉒	Expensive helium usage - helium
㉓	Lack hardware commonality
㉔	Propellant contamination
25	Side-mounted booster vehicles (multiple stage propulsion systems)

INTEGRATED PROPULSION MODULE IS FLEXIBLE

- **"Integrated" propulsion module is a single engine**
 - Meets wide range of thrust (1,00,000 - 4,000,000 Klbs) by adding or eliminating components
- **"Integrated" propulsion module is operationally efficient**
 - Simpler
 - More reliable
 - Greater engine-out capability
 - More robust
 - More operable (operationally efficient)
 - Lower cost
 - Lower weight

OEPSS CONCLUSION

- **Operations starts at design concept (TQM)**
- **Integrated design operationally efficient**
 - Substantially higher reliability and lower cost
 - New technology not required (enabling)
 - High flight rates and routine access to space
- **Other innovative propulsion concepts possible**

PROPULSION SYSTEMS OPTIONS-

CURRENT SYSTEMS

EXPENDABLE LAUNCH VEHICLE PROPULSION

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NASA Space Transportation
Propulsion Technology Symposium

EXPENDABLE LAUNCH VEHICLE PROPULSION

June 1990

P.N. Fuller, Chairman
COMSTAC Technology &
Innovation Working Group

CONTENTS

This presentation will review the current status of the U.S. ELV fleet, the international competition, and the propulsion technology of both domestic and foreign expendable launch vehicles. The ELV propulsion technology areas where research, development, and demonstration are most needed will be identified. These propulsion technology recommendations are based on the work performed by the Commercial Space Transportation Advisory Committee (COMSTAC), an industry panel established by the Department of Transportation.

Expendable Launch Vehicle Propulsion

Contents

- Introduction
- COMSTAC
- Domestic ELV Launch Fleet
- Foreign ELV Launch Fleet
- ELV Propulsion Systems
- ELV Propulsion Technology Needs
- Conclusions

INTRODUCTION

There have been extensive changes in America's space launch architecture since the Challenger tragedy occurred in January 1986. The major impact has been the revival of the U.S. Expendable Launch Vehicle (ELV) fleet in response to changes in National Space Launch policy. The NASA and the Air Force have adoped use of a "Mixed Fleet" of space launchers, and have prohibited the Shuttle Space Transportation System (STS) from competing for launch of commercial payloads. The availability of this diverse stable of launch systems has helped to assure access to space for critical payloads.

The foundation for a commercial launch industry has been established in the United States for the Delta, Atlas, and Titan III launch systems. The NASA and Air Force have provided a base for a commercial launch industry by long-range procurements of ELV launch services, and access to government facilities. U.S. industry has responded to legislation and enabling regulations by investments of private resources and funds.

However, international competition from government-subsidized launchers in Europe and Japan, and state-owned launch organizations in the non-market economies of the People's Republic of China (PRC) and the Soviet Union (USSR) threaten the survival of the U.S. commercial launch industry. The foreign launch systems enjoy competitive advantages due to government support for applied research and continued product development that need not be recovered in their pricing.

Similar support from NASA is needed to enhance the future competitiveness of the U.S. ELV industry. Near-term applied technology research aimed at cost reduction and product improvements to the current ELV fleet should be included in the NASA Research and Technology plans. In addition, long-term basic research is also needed to maintain parity with the new generation of foreign ELVs that will enter the market for commercial launch services in the mid and late 1990s.

Expendable Launch Vehicle Propulsion

Introduction

- **U.S. ELV launch fleet revived following 1986 STS-51 tragedy**
 - Change to "Mixed Fleet" national space launch policy
 - Need for assured access to space for critical payloads
- **Commercial ELV launch industry established for Delta, Atlas, Titan III**
 - Private industry responded to enabling legislation & regulations
 - Business base provided by NASA and Air Force procurements
- **International competition threatens U.S. commercial launch services**
 - Government supported launch industries in Europe & Japan
 - State-owned launch systems from non-market economies in PRC & USSR
- **NASA basic and applied research funding needed for ELVs**
 - Near-term improvement of current ELV propulsion
 - Long-term basic propulsion research

COMMERCIAL SPACE TRANSPORTATION ADVISORY COMMITTEE

The Commercial Space Transportation Advisory Committee (COMSTAC) is an advisory group to the Office of Commercial Space Transportation (OCST) of the Department of Transportation (DoT). The OCST reports to the Secretary of Transportation, the Honorable Samuel K. Skinner, who is a member of the National Space Council (NSpC). The Director of OCST is Stephanie Lee-MIller.

The objective of the COMSTAC is to promote U.S. commercial space transportation by acting as an advocate for private industries involved in providing space transportation goods and services. COMSTAC provides, thru the OCST, industry views on space transportation policies, regulations, and procedures. The chairman of COMSTAC is Dr. Alan Lovelace of the General Dyanmics Corporation.

COMSTAC consists of a full committee of 23 appointed members from small, medium, and large corporations representing space transportation suppliers and users. The committee is organized into five (5) Working Groups:

- Technology & Innovation Working Group
- Infrastructure Working Group
- Insurance & Risk Management Working Group
- International Competition & Cooperation Working Group
- Procurement Working Group

Each working group, headed by a member of the full committee, is the focus of COMSTAC efforts on specific issues and areas relevant to space transportation.

The Technology & Innovation Working Group, chaired by Mr. Paul N. Fuller of Rocketdyne, is responsible for identifying and prioritizing technology needs (including propulsion technology) for commercial ELVs. The working group has been chartered to review and advise on the NASA Component Technology Plans, and to work on a long-range plan for industry-government cooperation to develop the next generation of U.S. commercial expendable launch vehicles.

Expendable Launch Vehicle Propulsion

Commercial Space Transportation Advisory Committee (COMSTAC)

- **Industry advisory committee to Office of Commercial Space Transportation (OCST) of Department of Transportation (DoT)**
- **Objective: Promote U.S. space transportation industry**
 - Industry views to Transportation Secretary and National Space Council
 - Review & recommendations on space policies & procedures
- **Organization: Committee of the whole and Working Groups**
 - Technology & Innovation • Procurement • Infrastructure
 - International Competition & Coop. • Insurance & Risk Management
- **Charter: Technology & Innovation Working Group**
 - Define technology needs for U.S. space transportation industry
 - Review & advise on NASA Component Technology Plan
 - Develop long-range plan for industry/government cooperation

COMSTAC TECHNOLOGY & INNOVATION WORKING GROUP

WORKING GROUP CHARTER

The Technology & Innovation Working Group, chaired by Mr. Paul N. Fuller of Rocketdyne, is responsible for identifying and prioritizing technology needs (including propulsion technology) for commercial ELVs. The working group has been chartered to review and advise on the NASA Component Technology Plans, and to work on a long-range plan for industry-government cooperation to develop the next generation of U.S. commercial expendable launch vehicles.

Expendable Launch Vehicle Propulsion

COMSTAC Technology & Innovation Working Group

Working Group Charter

- Offer advice on NASA's Component Technology Program by defining areas or programs which offer the greatest payoff in expenditure of Research & Technology funds toward assuring future world-wide competitiveness of the U.S. space transportation industry.

- Develop a long range plan for a joint industry/government cooperative project to develop next generation U.S. commercial ELVs. Include in the plan integration of NASA's Component Technology Program, ALS Technology Programs, and the President's Space Exploration Initiative.

COMSTAC TECHNOLOGY & INNOVATION WORKING GROUP
WORKING GROUP MEMBER ORGANIZATIONS

The FY90 members of the Technology & Innovation working group represent 12 U.S. corporations involved in supplying goods and services to the commercial space launch industry. The membership represents new and emerging industries as well as large, established organizations that have been involved in space launch systems for over 35 years.

Expendable Launch Vehicle Propulsion

COMSTAC Technology & Innovation Working Group

Working Group Member Organizations:

- **American Rocket Company (AMROC)**
- **American Telephone & Telegraph Corporation**
- **Boeing Helicopters**
- **GenCorp, Inc. - Aerojet TechSystems**
- **General Dynamics Corporation - Commercial Launch Services**
- **International Technology Underwriters (INTEC)**
- **Martin Marietta Corp. - Commercial Titan Inc.**
- **McDonnell Douglas Corp. - Space Systems Company**
- **Rockwell International - Rocketdyne Division**
- **Space Services, Inc. of America**
- **United Technologies Corporation - Pratt & Whitney Division**
- **United Technologies Corporation - United Space Boosters, Inc.**

DOMESTIC ELV LAUNCH FLEET

The domestic ELV launch fleet consists of the following launch systems:

SYSTEM	SUPPLIER	USERS
• Titan II	Martin Marietta Corporation	Military, NASA
• Delta II	McDonnell Douglas Space Systems	Military, NASA, Commercial
• Atlas I, II	General Dynamics Space Systems	Military, NASA, Commercial
• Titan III	Martin Marietta Corporation	NASA, Commercial
• Titan IV	Martin Marietta Corporation	Military, Government

With the exception of the Titan II, production has resumed for construction of all new ELV components and flight hardware. The Delta, Atlas, and Titan III launch systems are available for commercial payloads. Titan II and Titan IV are used for military and/or other government launches.

The domestic fleet of ELVs were derived from ballistic missiles and government launchers developed in the 1950's and 1960's. The current launch system configurations are the result of evolutionary, incremental uprates and improvements made to the propulsion systems, vehicle structures, avionics, manufacturing processes, and launch facilities. The systems used in the U.S. ELV fleet, including propulsion subsystems, are mature, flight-proven designs; however, commercial application and low cost were not initial design considerations.

The private sector has made significant investments in ELV launch systems on the assumption that commercial markets will develop. Firms such as General Dynamics have invested several hundred million dollars in facilities, start-up costs, and quantity orders based on assumed capture of targeted segments of the commercial launch services market. Total private industry cash flow commitment and capital investment is estimated to exceed $500M. Government contracts have benefited from these investments through lower unit costs for launches of government payloads.

However, private industry cannot afford the additional investment in the non-recurring costs needed to develop new launch systems to meet the competitive challenges of foreign launch vehicles in the mid-1990s.

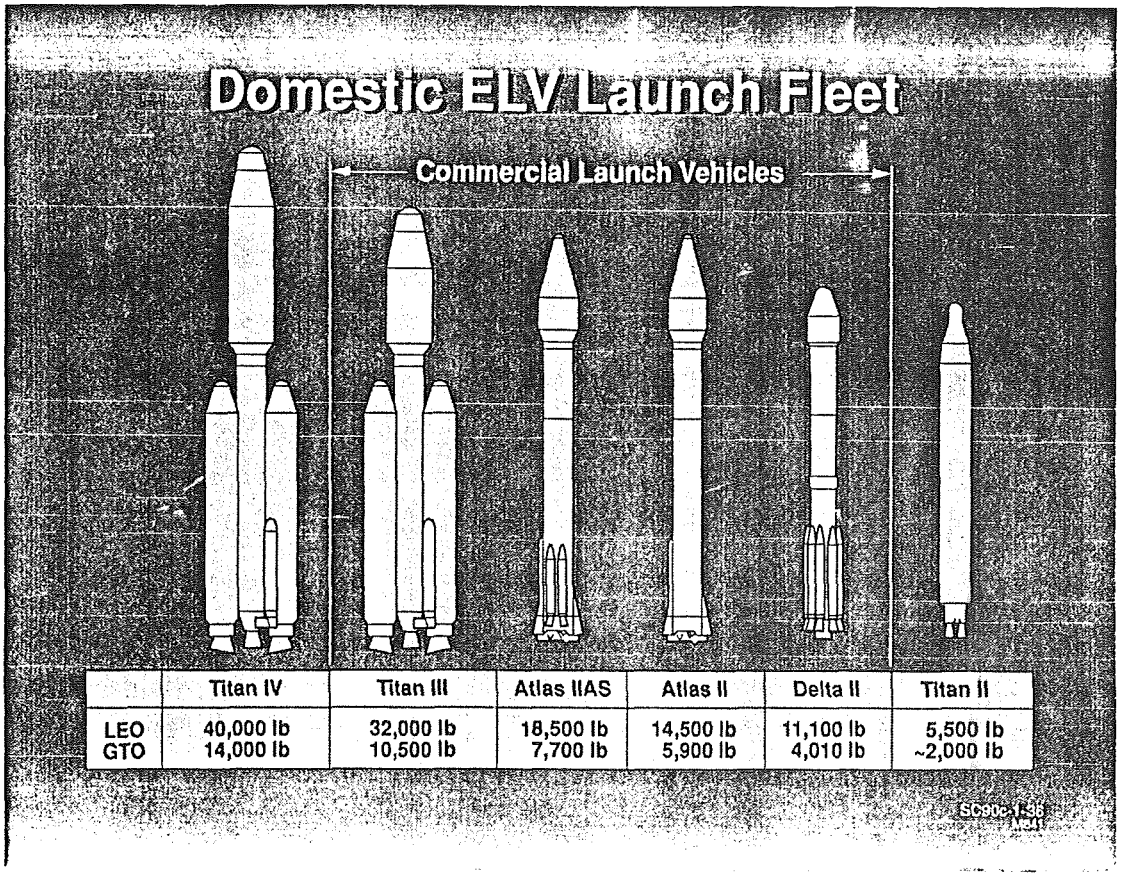
Expendable Launch Vehicle Propulsion

Domestic ELV Launch Fleet

- **Production and launches resumed for U.S. ELV fleet**
 - Titan II — Martin Marietta — Military
 - Delta II — McDonnell Douglas — Military, NASA, Commercial
 - Atlas II — General Dynamics — Military, NASA, Commercial
 - Titan III — Martin Marietta — NASA, Commercial
 - Titan IV — Martin Marietta — Military, NASA
- **Derived from ballistic missiles & government space launchers**
 - Propulsion systems are mature, flight proven designs
 - Current configurations - incremental uprates & improvements
 - Commercial application & low cost not initial design considerations
- **Private sector made significant investment in ELV launch systems**
 - Start-up costs and quantity orders of materials & systems
 - Cannot afford non-recurring development costs of new systems

DOMESTIC ELV LAUNCH FLEET

The U.S. Expendable Launch Vehicle (ELV) fleet is shown on the opposite page with their payload capabilities to Low-Earth Orbit (LEO) and Geosynchronous Transfer Orbit (GTO). The Delta II, Atlas II, and Titan III are competitors for commercial launch services. Some order has been established in the domestic launch service market. Each of the launch systems has its own market niche where it has a competitive advantage in either payload capability or launch price.



	Titan IV	Titan III	Atlas IIAS	Atlas II	Delta II	Titan II
LEO	40,000 lb	32,000 lb	18,500 lb	14,500 lb	11,100 lb	5,500 lb
GTO	14,000 lb	10,500 lb	7,700 lb	5,900 lb	4,010 lb	~2,000 lb

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FOREIGN ELV LAUNCH FLEET

The international competition for commercial launch services is fierce. Arianespace is the industry leader, currently capturing about 50% of the market. Arianespace is the launch services marketing organization for the Ariane family of vehicles developed by the European Space Agency (ESA), a multi-national consortium. Arianespace enjoys a competitive advantage in the international launch services market. Its launch pricing is based only on recovery of recurring cost, and its large backlog of commercial and captive ESA payloads enables flexibility in manifesting. Furthermore, with ESA support, Arianespace has also demonstrated it is able to develop and market new launch vehicles in a short period of time. The Ariane 4 was recently introduced to replace the 5-year old Ariane 3 system. Continuing non-recurring development support from ESA for Ariane 4 is estimated to be $50M/year. An all-new Ariane 5 is being developed with ESA funds (~ $5B/year) for introduction in 1995.

State-owned launch systems from non-market economies are recent entrants into the market for commercial launch services. The People's Republic of China's (PRC's) Long March family of launch vehicles have captured launch contracts for the Asiasat, Aussat, and Arabsat spacecraft. The Soviet Union is also poised to enter the commercial launch services market with its Proton and Zenit launch vehicles.

The PRC and USSR enjoy a competitive advantage vis-a-vis private firms from Western market economies by being able to price launch services independently of costs. This ability to arbitrarily price is the major threat to the future survival and growth of the U.S. commercial space launch industry.

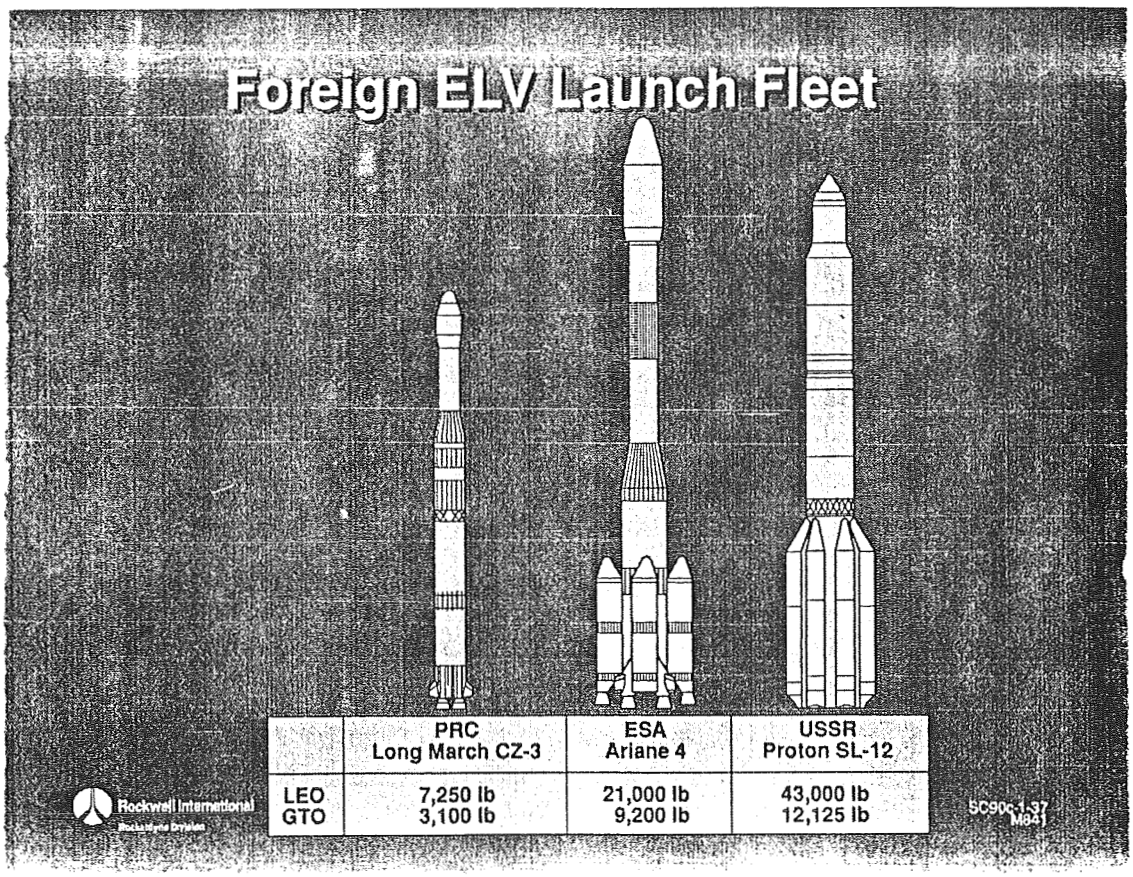
Expendable Launch Vehicle Propulsion

Foreign ELV Launch Fleet

- **Fierce international competition for launch service contracts**
- **Ariane is industry leader - 50% of commercial market**
 - European government consortium (ESA) supported development
 - Captive ESA payloads enables flexibility in manifesting
 - Converted from Ariane 3 (1983 - 1989) to Ariane 4 (1988 - present)
- **Arbitrary pricing competition from non-market economies**
 - Long March CZ-3 — People's Republic of China (PRC)
 - Proton SL-12 & Zenit SL-16 — Soviet Union (USSR)
- **Near-term threats from ELVs designed for commercial market**
 - Long March CZ-2E & CZ-3A — PRC — 1991
 - H-II — Japan — 1993
 - Ariane 5 — ESA — 1995

FOREIGN ELV LAUNCH FLEET

Foreign Expendable Launch Vehicles (ELV) competing against U.S. ELVs for commercial launch services are shown on the opposite page. Note that their LEO and GTO payload capabilities are equivalent to U.S. launchers. Since technical capability are equivalent, price and market access are the key competitive issues. Launch system reliability (as indicated by insurability) has not yet been a discriminating competitive feature.



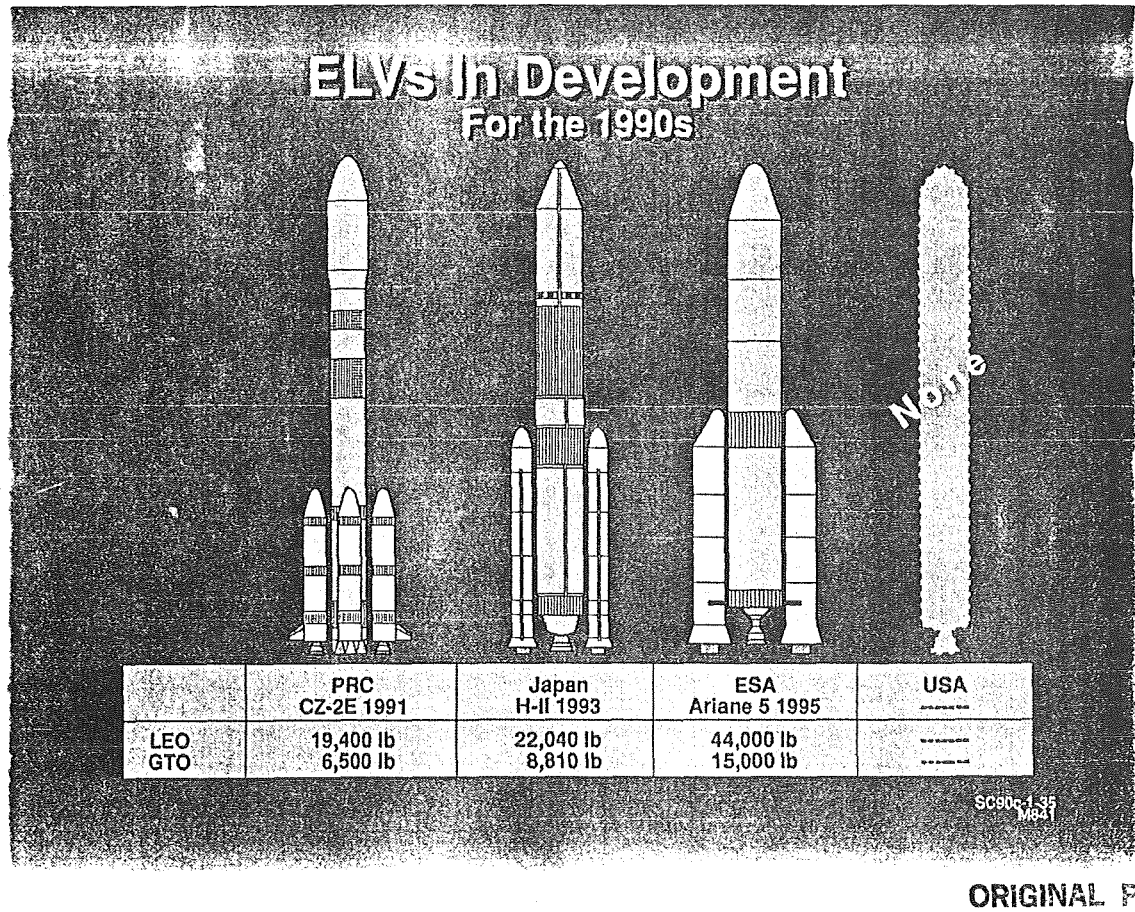
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ELV's IN DEVELOPMENT FOR THE 1990's

The current competitive environment will become even more difficult for the U.S. commercial space launch industry in the mid-1990's when new launch systems from the PRC, Japan, and ESA become operational. The PRC is currently developing the CZ-2E (shown) an uprated version of CZ-2 launch system. The National Aeronautics and Space Development Agency (NASDA) of Japan is funding development of the H-II launch system that utilizes all LOX/H2 propulsion systems. Similarly , ESA is funding development of the LOX/H2 Ariane 5 launch system.

Each of these systems are being designed specifically for the commercial segment of the space launch market. Further, the H-II and Ariane 5 are based on current state-of-art in propulsion, avionics, materials, structures, manufacturing, and launch operations. The Ariane 5 is designed to reduce the price of payload weight to orbit by 40% compared to the Ariane 4.

The U.S. has no comparible launch system under development at the current time.



	PRC CZ-2E 1991	Japan H-II 1993	ESA Ariane 5 1995	USA ------
LEO	19,400 lb	22,040 lb	44,000 lb	------
GTO	6,500 lb	8,810 lb	15,000 lb	------

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U.S. ELV PROPULSION SYSTEMS

The propulsion systems in the current fleet of U.S. expendable launch vehicles were designed for ballistic missiles and government space launchers in the mid-1950's and early 1960's. The liquid rocket engines for the Thor, Atlas, and Titan launch vehicles were developed specifically for their intermediate and intercontinental range ballistic missile missions under Air Force contracts awarded beginning in 1954. The Delta launch vehicle utilizes engine hardware designed for the H-1 engine used in the NASA's Apollo program.

These engine are mature designs, and have an outstanding records of flight success. Extensive production and launch history databases exists for these engine systems:

Engine System	Propellant	Systems Delivered	Launches
• Thor/Delta Engines	LOX/RP	>610	>550
• Atlas Engines	LOX/RP	>640	>490
• Titan Stage I Engines	NTO/UDMH	>310	>240

The current ELV propulsion system configurations are a result of continuous evolutionary performance improvements made to the original engine designs:

	Original	Current
• Delta Main propulsion thrust:	135,000 lb	207,000 lb
• Atlas Booster propulsion thrust:	270,000 lb	423,500 lb
• Titan Stage I propulsion thrust:	430,000 lb	546,000 lb

Propulsion system modifications were made over the years to satisfy specific mission requirements, and were funded incrementally to minimize cost and expedite schedule.

Although the propulsion systems for the current ELV fleet have outstanding heritages of flight reliability, the designs are based on requirements and techniques reflecting the state of the art of the 1950's and 1960's. Certain engine components had been out of production for over 20 years. In the recent production resumption, modern manufacturing processes and procedures have been applied to reduce cost and improve quality. However, since the systems were not originally designed specifically for low cost nor commercial applications, the benefits of this approach have been limited. Furthermore, the designs are operating near their inherent design limits due to the numerous upratings performed in the past.

Expendable Launch Vehicle Propulsion

U.S. ELV Propulsion Systems

- **Designed for ballistic missiles & government space launchers**
 - Thor/Atlas/Titan engines: Initial production 1955 - 1960
 - Delta (H-1) engines: Initial production 1960 - 1964
- **Mature designs with outstanding flight success histories**

	Engines delivered	Launches
• Thor/Delta engines:	610+	550+
• Atlas engines:	640+	490+
• Titan* Stage I engines:	310+	240+

- **Continuous evolutionary performance improvements made; but hardware near design limits**

	Original	Current
• Delta main propulsion thrust:	135,000 lb	207,000 lb
• Atlas booster propulsion thrust:	270,000 lb	423,500 lb
• Titan* Stage I propulsion thrust:	430,000 lb	546,000 lb

* Titan II to Titan IV storable propellant engine systems

FOREIGN COMMERCIAL ELV PROPULSION SYSTEMS

A formidable array of propulsion systems are utilized in foreign launch vehicles. The Ariane 4 uses the Viking storable propellant engines as main propulsion in the core vehicle, as well as a combination of liquid and solid propellant strap-on boosters. These engines were designed and developed in the 1970's. An efficient LOX/H2 upper stage engine, developed in the early 1980's, is utilized for transfer orbit insertion. Commercial launches are conducted from modern vehicle assembly and launch facilities located near the equator in French Guyana. Development costs of all Ariane launch facilities, launch vehicles, and propulsion systems are funded by the ESA consortium.

Storable propellant booster and LOX/H2 upper stage engines are also used in the PRC's Long March vehicles offered for commercial spacecraft launches. The Long March vehicles are based on military launch systems, and the entire launch service (propulsion system and vehicle production, payload integration, and launch operations) is conducted as a state-owned industry.

The Soviet Union's Proton launch vehicle is powered by storable propellant booster engines, and a LOX/RP upper stage engine. The storable propellant engines are of advanced design, and operate at higher chamber pressures than comparable ELV systems in the U.S. or Europe. The Proton is only one of 9 military ELV launch systems available in the Soviet space launch fleet. Indications are that other Soviet launch systems will be offered on the commercial launch market in the near future.

The Japanese are in the final development stages of their H-II expendable launch vehicle. The Japanese NASDA funds all propulsion, vehicle, and launch facility development activities. The LOX/H2 LE-5 upper stage engine has flow successfully in 3 missions on the current H-I launch vehicle. The LOX/H2 LE-7 main propulsion system is currently undergoing development testing. The H-II is specificially designed for non-military applications, and is scheduled for initial launch in 1993.

Expendable Launch Vehicle Propulsion

Foreign Commercial ELV Propulsion Systems

- **ESA Ariane 4 (present) and Ariane 5 (1995)**
 - Ariane 4: Storable propellant booster & LOX/H2 upper stage engines
 - Ariane 5: LOX/H2 booster & upper stage engines
 - Propulsion developed by ESA for low-cost space applications
- **PRC Long-March CZ-3 (present), CZ-2E (1991), & CZ-3A (?)**
 - Storable propellant boosters & LOX/H2 upper stage engines
 - State-sponsored commercial launchers based on military systems
- **USSR Proton SL-12 (present) and Zenit SL-16 (?)**
 - Storable propellant booster & LOX/RP upper stage engines (Proton)
 - LOX/RP booster and upper stage engines (Zenit)
 - Advanced technology, high chamber pressure engines
 - One of 9 ELV military space launch systems
- **Japan H-II (1993)**
 - LOX/H2 booster and upper stage engines
 - Cryogenic engine technology equivalent to U.S. SSME & RL-10
 - Propulsion developed by NASDA for non-military applications

ELV PROPULSION TECHNOLOGY NEEDS

Public Law 100-657 "Commercial Space Launch Act Admendments of 1988" directed that NASA, in consultation with the U.S. space launch industry, design a research and technology program for launch system components aimed at the development of higher performance and lower cost launch vehicles for commercial and government payloads to ensure development of a competitive domestic ELV industry."

The COMSTAC Technology & Innovation Working Group has been tasked to identify and prioritize technologies needed to enhance ELV competitiveness, and to advise on the NASA Component Technology Plan.

Beginning in 1989, COMSTAC has provided inputs to the NASA Component Technology Plan. The Working Group is currently completing its report on ELV technology needs in the areas of propulsion, avionics, structures (& materials), production processes, and launch operations.

The list of technologies needed in the area of ELV Propulsion was compiled by the Working Group independently of the NASA plan. A preliminary version of the list is shown in the following charts, and is divided into the technologies needed to support:

- Liquid Propulsion
- Solid Propulsion
- Hybrid Propulsion.

The areas identified have been prioritized based on a consensus of the Working Group members. The final report of the Working Group will be submitted to the full COMSTAC committee before the end of the fiscal year.

Expendable Launch Vehicle Propulsion

ELV Propulsion Technology Needs

- **COMSTAC Technology & Innovation Working Group**
 - Identify technologies needed to enhance ELV competitiveness
 - Advise on NASA Technology Plans as mandated by 100th Congress
 - Propulsion, Avionics, Structures, Production, Launch Operations

- **Specific ELV Propulsion technologies identified and prioritized**
 - Liquid Propulsion
 - Solid Propulsion
 - Hybrid Propulsion

- **NASA/OAET Component Technology Plan reviewed**
 - Generally in agreement with ELV Plan - Propulsion
 - Needs more near & mid-term focus for commercial launch industry
 - Develop & demonstrate technologies to enhance current ELVs
 - Support development of new family of ELVs

NASA COMPONENT TECHNOLOGY PLAN

The COMSTAC Technology & Innovation Working Group reviewed the NASA Component Technology Plan submitted to the OMB in March 1990. In general, the Working Group agreed with the NASA Plan in the area of Propulsion. However, we felt that the NASA Plan needed to focus more on near-term (1 - 5 years) and mid-term (5 - 7 years) technology development activities. The Working Group believes that applications from these technology programs are going to be required in the mid to late 1990's to remain competitive with the foreign ELV competition.

The Working Group also felt that the NASA Plan should include tasks that involve the development and demonstration of technologies to enhance the *current* fleet of ELVs. These near-term activities could be the development of prototypes of cost reduction product improvements, demonstrations of significant performance enhancements concepts, and/or applied technology demonstrations of propulsion system components and subsystems.

Finally, the consensus of the Working Group is that the overall NASA Plan should recognize the need for government support to develop a new family of ELVs that have commercial applicability. Current NASA and Air Force plans focus on manned or advanced launch systems that provide heavy lift capability. Heavy lift systems have little commercial applicability in the foreseeable future, and the family of advanced launch vehicles should include configuations that can down-sized for commercial payloads.

Expendable Launch Vehicle Propulsion

NASA Component Technology Plan

- **Background:**

 Section 10 of Public Law 100-657 "Commercial Space Launch Act Amendments of 1988" directed that NASA:

 "In consultation with representatives of the Space Launch Industry, design a program for the support of research into launch systems component technologies, for the purpose of developing higher performance and lower cost U.S. launch vehicle technologies and systems available for the launch of commercial and government spacecraft into orbit.."

- **Purpose:**

 "To ensure the successful development of a competitive domestic expendable launch vehicle (ELV) industry.."

Expendable Launch Vehicle Propulsion

Liquid Propulsion Technology Needs

ITEM	PRIORITY
• Low cost liquid booster engines	
A. New LOX/H2 engine	1
B. Evolutionary LOX/RP engine	1
• Advanced low cost LOX/H2 upper stage engine	
A. 30-50K lb thrust	1
B. 100-200K lb thrust	2
• Improved hydrocarbon propellant derivative engines & components	2
• Leak-free engine propulsion & pressurization subsystems (joints, tubing, ducts)	2

Priority 1: Highest payoff - must do. Priority 2: Should do.
Priority 3: Good to do.

Expendable Launch Vehicle Propulsion

Liquid Propulsion Technology Needs (continued)

ITEM	PRIORITY
• Automated fluid, mechanical, and propulsion subsystem checkout	2
• Liquid air cycle engine (LACE)	2
• Pressure-fed propulsion subsystem technologies (cryo helium storage, autogen. pressurization systems)	2
• Booster recovery and reuse technologies	2
• Electronic pressure controllers	2
• Low-cost pressure fed engine	3
• LOX/H2 reaction control system (RCS) and ΔV system	3

Priority 1: Highest payoff - must do. Priority 2: Should do.
Priority 3: Good to do.

Expendable Launch Vehicle Propulsion

Solid Propulsion & Hybrid Propulsion Technology Needs

ITEM	PRIORITY
• **Low cost filament wound motorcases**	**2**
• **Castable ablative nozzles**	**2**
• **Hybrid propulsion strap-on booster**	**2**
• **Clean burning solid motors**	**3**

Priority 1: Highest payoff - must do. Priority 2: Should do.
Priority 3: Good to do.

CONCLUSIONS

The U.S. no longer leads in ELV propulsion system technology nor in operational launch systems. The fleet of domestic ELVs are powered by propulsion systems that are reliable, but of aging design. Private industry cannot afford the investment in non-recurring development costs for new low-cost commercial launch systems that will be needed in the mid-1990s to compete against the modernized, low-cost foreign systems that will become operational.

NASA technology support is needed to regain leadership in space transportation. This includes near-term development and demonstration activities of propulsion technologies and applications that reduce cost or enhance the capabilities of the current fleet of domestic ELVs. It is also needed in basic propulsion technologies and vehicle development for a new family of low-cost ELVs with commercial applicability.

The COMSTAC industry advisory group has identified the propulsion technologies needed to enhance the future competitiveness of the domestic space launch industry. It stands ready and willing to support NASA and its plans for propulsion technology development.

A strong commercial launch industry will benefit the future of the U.S. It will reduce launch costs to the government, provide assured access to space for critical payloads, and contribute to economic growth and international trade in the 21st Century.

Expendable Launch Vehicle Propulsion

Conclusions

- **U.S. no longer leads in ELV propulsion systems**
 - Reliable but aging U.S. designs
 - Competing against modern low-cost foreign systems
- **NASA technology support needed to regain leadership**
 - Near-term development & demonstration of ELV supporting technologies
 - Basic research & technologies for new family of low-cost ELVs
- **COMSTAC ready to support NASA**
 - Identify ELV Industry needs and priorities
 - Promote NASA's budget and programs
- **A strong commercial launch industry benefits U.S.**
 - Reduces launch costs to government
 - Assured access to space for critical payloads
 - Economic growth & trade balance considerations

SHUTTLE PROPULSION SYSTEMS

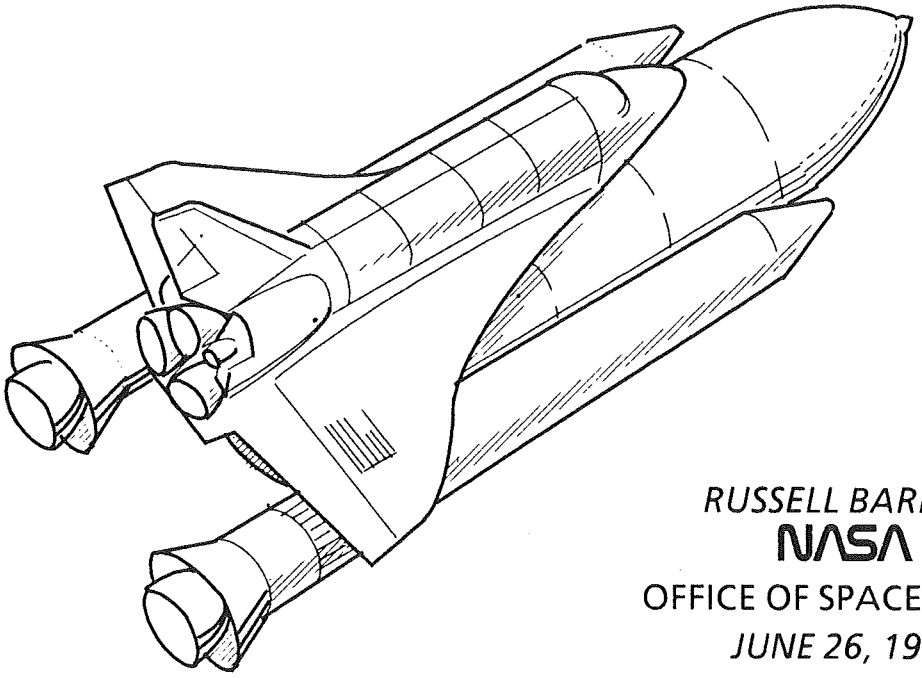
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SPACE SHUTTLE PROPULSION SYSTEMS

SPACE TRANSPORTATION TECHNOLOGY SYMPOSIUM

PENNSYLVANIA STATE UNIVERSITY



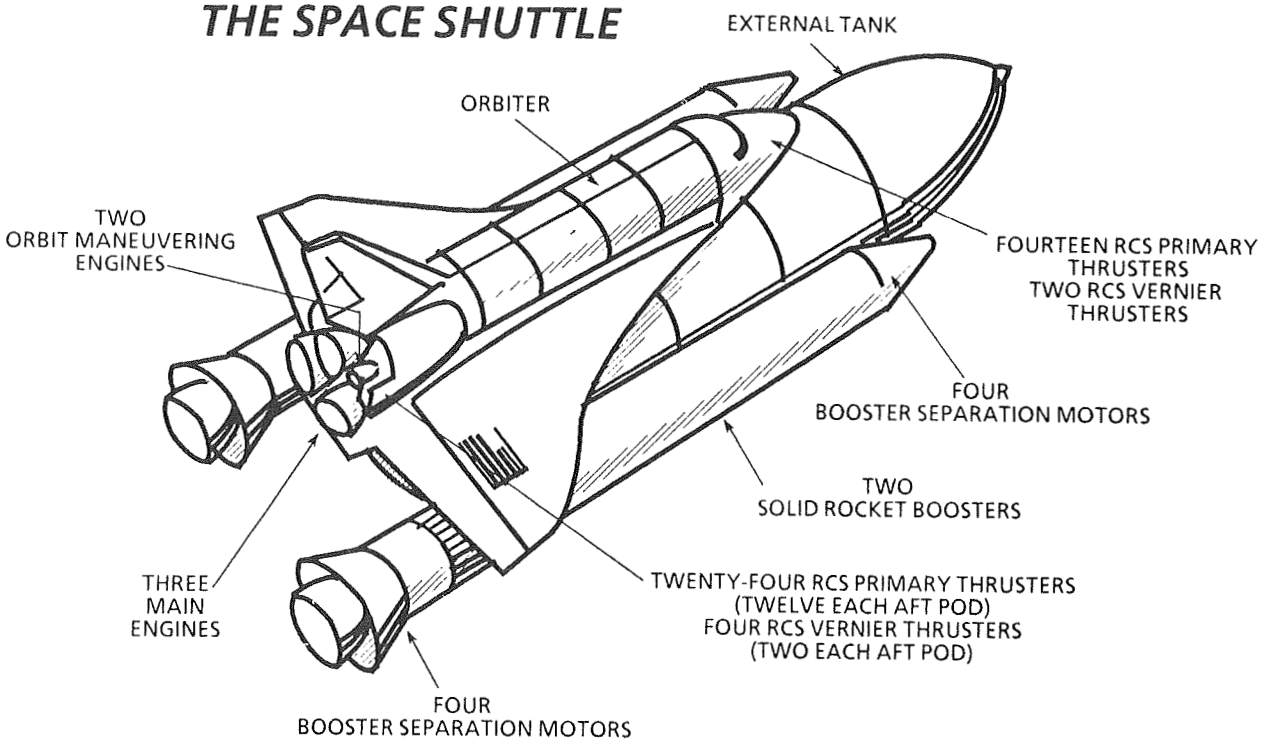
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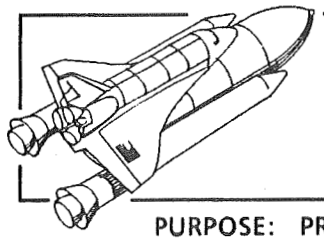
NASA

OFFICE OF SPACE FLIGHT

JUNE 26, 1990

THE SPACE SHUTTLE

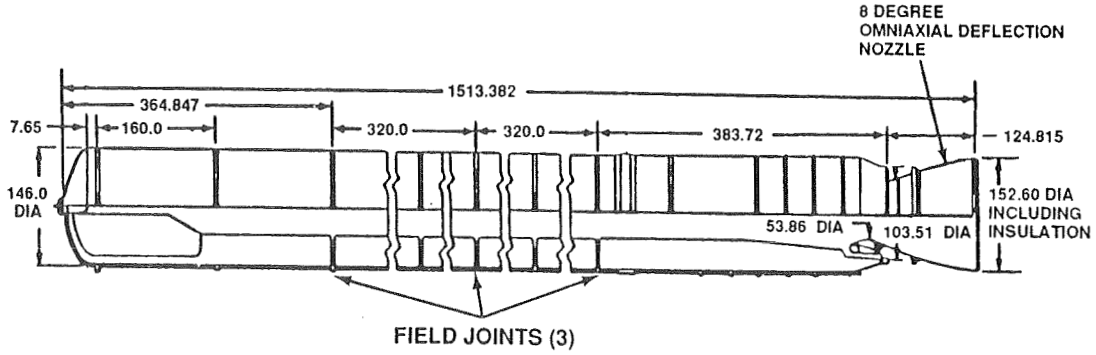


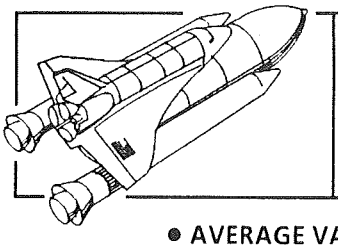


REDESIGNED SOLID ROCKET MOTOR
Four Segment Design

PURPOSE: PROVIDES PROPULSIVE THRUST FROM LIFTOFF THROUGH THE FIRST 123 SECONDS OF FLIGHT

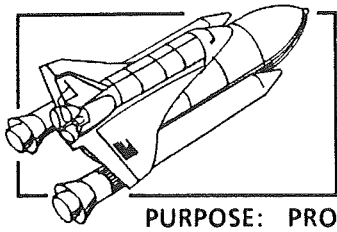
SUPPLIER: THIOKOL CORP., WASATCH, UTAH





RSRM DESIGN PARAMETERS

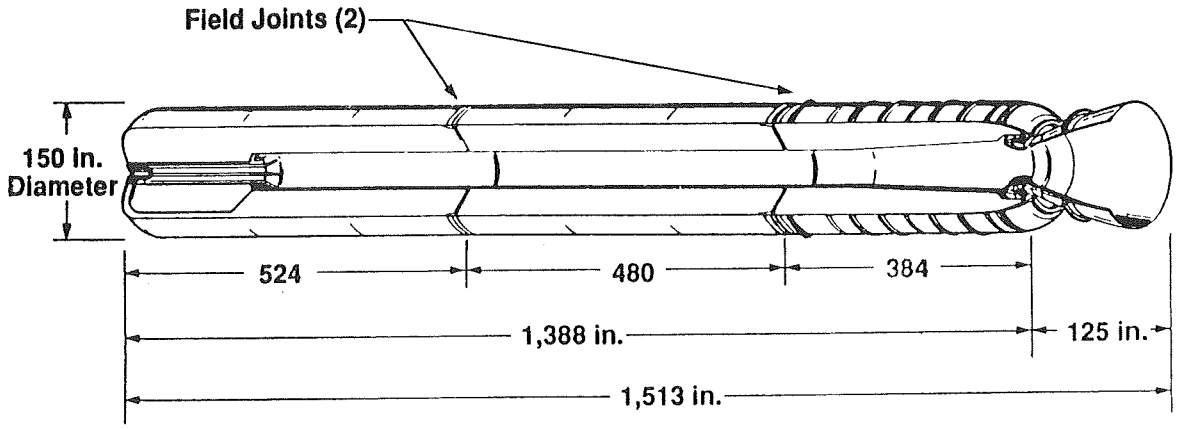
• AVERAGE VACUUM THRUST (WEB TIME)	2,590,000 LBS
• SPECIFIC IMPULSE (VACUUM)	267.9 SEC
• AREA RATIO (A_e/A_t)	7.72
• AVERAGE CHAMBER PRESSURE	625 PSIA
• ACTION TIME	123.4 SEC
• MOTOR WEIGHT	1,255,978 LBS
• PROPELLANT WEIGHT	1,107,169 LBS
• MASS FRACTION	0.882
• INERT WEIGHT:	
CASE	98,740 LBS
NOZZLE	23,965 LBS
• PROPELLANT TYPE	PBAN
• BURN RATE (@625 PSIA)	0.368 IN/SEC
• THRUST VECTOR CONTROL	FLEX BEARING
• CASE MATERIAL	D6AC STEEL
• INSULATION MATERIAL	ASBESTOS/NBR

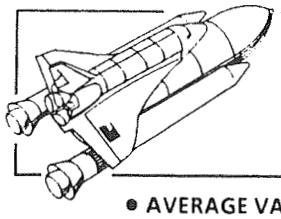


ADVANCED SOLID ROCKET MOTOR
Three Segment Design

PURPOSE: PROVIDES PROPULSIVE THRUST FROM LIFTOFF THROUGH THE FIRST 134 SECONDS OF FLIGHT

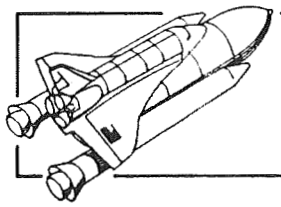
SUPPLIER: LOCKHEED MISSILES & SPACE COMPANY, SUNNYVALE, CA.





ASRM DESIGN PARAMETERS

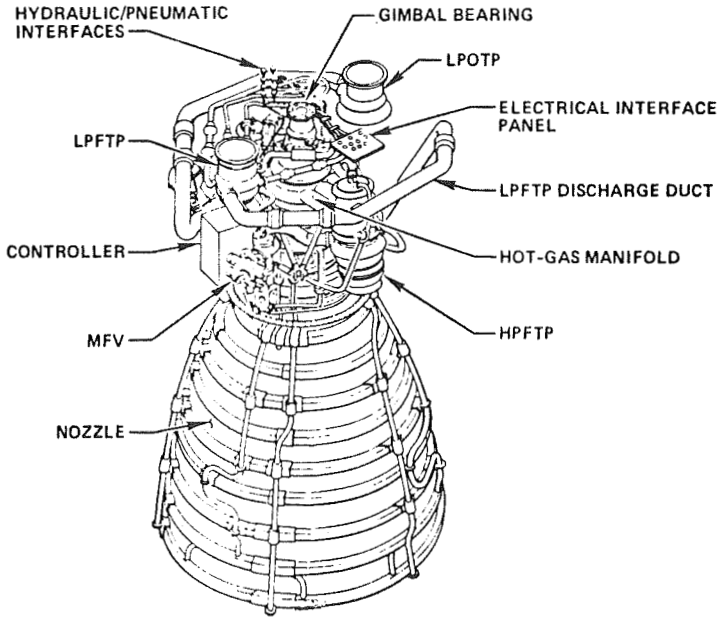
• AVERAGE VACUUM THRUST (WEB TIME)	624,031 LBS
• SPECIFIC IMPULSE (VACUUM)	70.3 SEC
• AREA RATIO (A_e/A_t)	7.54
• AVERAGE CHAMBER PRESSURE	633 PSIA
• ACTION TIME	134.1 SEC
• MOTOR WEIGHT	1,345,807 LBS
• PROPELLANT WEIGHT	1,205,807 LBS
• MASS FRACTION	8.96
• INERT WEIGHT:	
CASE	97,419 LBS
NOZZLE	18,947 LBS
• PROPELLANT TYPE	HTPB
• BURN RATE (@625 PSIA)	0.345 IN/SEC
• THRUST VECTOR CONTROL	FLEX BEARING
• CASE MATERIAL	9 Ni-4 Co-0.3C
• INSULATION MATERIAL	KEVLAR-GLASS-EPDM

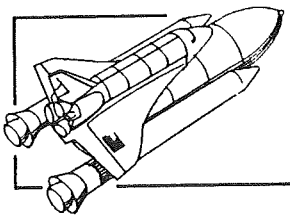


SPACE SHUTTLE MAIN ENGINE

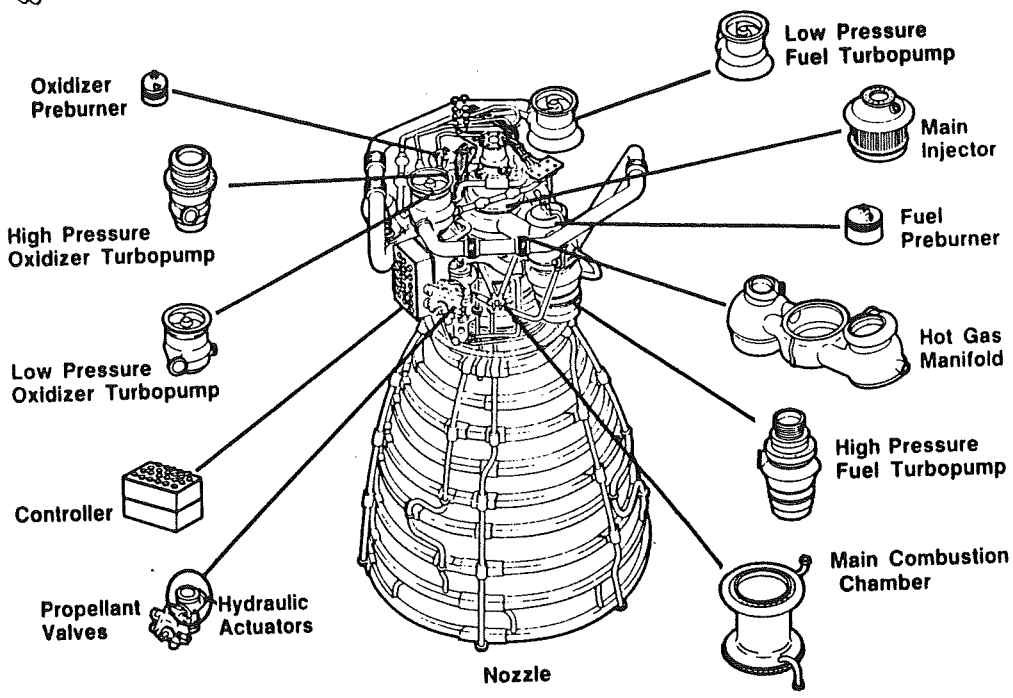
PURPOSE: PROVIDE PROPULSIVE THRUST FROM LIFTOFF TO ORBIT

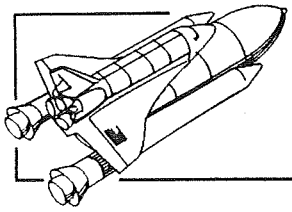
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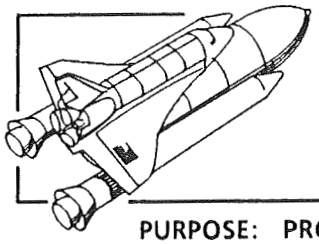
SSME COMPONENTS





MAIN ENGINE PARAMETERS

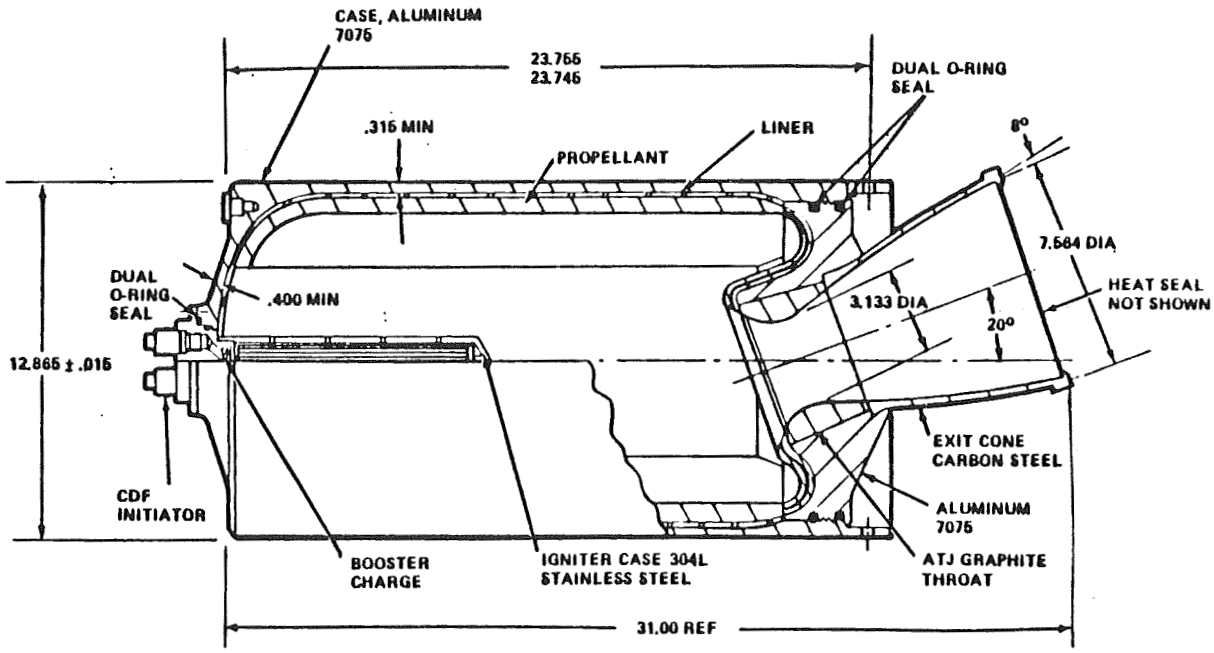
• PROPELLANTS	OXYGEN/HYDROGEN
• RATED POWER LEVEL (RPL) 100%	470,000 LBS
• FULL POWER LEVEL (FPL) 109%	512,300 LBS
• MINIMUM POWER LEVEL (MPL) 65%	305,500 LBS
• THROTTLE RANGE	65% TO 109% (1% Increments)
• CHAMBER PRESSURE	3200 PSIA
• MIXTURE RATIO	6.03 : 1
• SPECIFIC IMPULSE	453.5 SEC
• FLOW RATES: OXYGEN HYDROGEN	 973 LB/SEC 161 LB/SEC
• WEIGHT	7,000 LBS
• DESIGN LIFE	27,000 SEC 55 STARTS
• FULL POWER LEVEL	14,000 SEC
• OVERALL HEIGHT	14 FEET
• NOZZLE DIAMETER @ EXIT	7.5 FEET

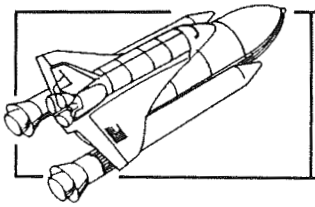


SRB BOOSTER SEPARATION MOTOR

PURPOSE: PROVIDES PROPULSIVE THRUST TO SEPARATE SRBs FROM THE ORBITER AND EXTERNAL TANK

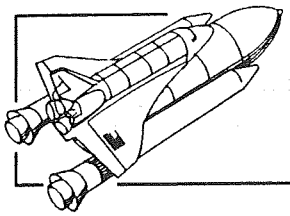
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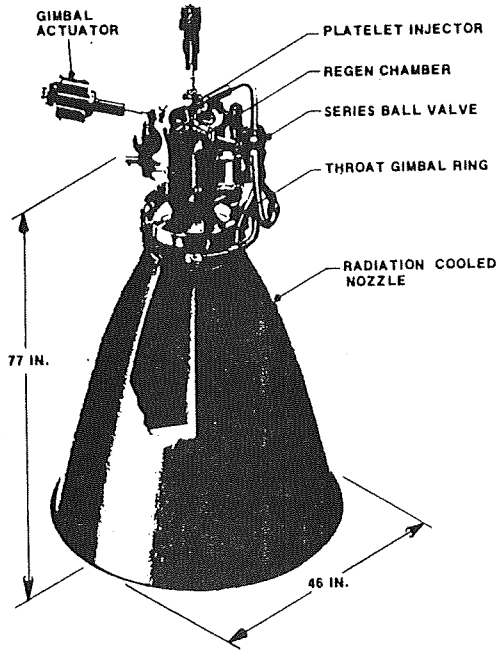


BSM DESIGN PARAMETERS

• AVERAGE VACUUM THRUST	20,050 LBS
• AREA RATIO	5.8
• AVERAGE CHAMBER PRESSURE	2221 PSIA
• ACTION TIME	0.805 SEC
• TOTAL IMPULSE	15,000 LB - SEC
• MOTOR WEIGHT	167 LBS
• PROPELLANT TYPE	HTPB
• CASE MATERIAL	7075 AL

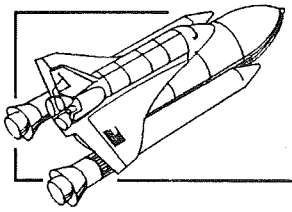


OMS ENGINE



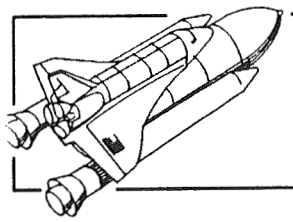
PURPOSE: PROVIDES PROPULSIVE THRUST FOR ORBIT INSERTION, ORBIT CIRCULARIZATION, ORBIT TRANSFER, RENDEZVOUS, DEORBIT, AND LAUNCH ABORT

SUPPLIER: AEROJET PROPULSION DIVISION; SACRAMENTO, CA.



OMS ENGINE DESIGN PARAMETERS

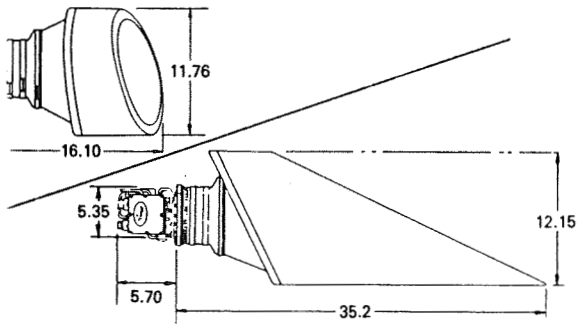
• PROPELLANTS	MMH/N_2O_4
• THRUST (VACUUM)	6,000 LBS
• NOMINAL SPECIFIC IMPULSE	313.2 SEC
• CHAMBER PRESSURE	125 PSIA
• MIXTURE RATIO	1.65
• EXPANSION RATIO	55:1
• FLOW RATES	
FUEL	11.93 LB/SEC
OXIDIZER	7.23 LB/SEC
• DRY WEIGHT	297 LBS
• LIFE	100 MISSIONS
	1000 STARTS
	15 HOURS CUM. FIRING
• GIMBAL CAPABILITY	
PITCH	± 6 DEG
YAW	± 7 DEG



RCS PRIMARY AND VERNIER THRUSTERS

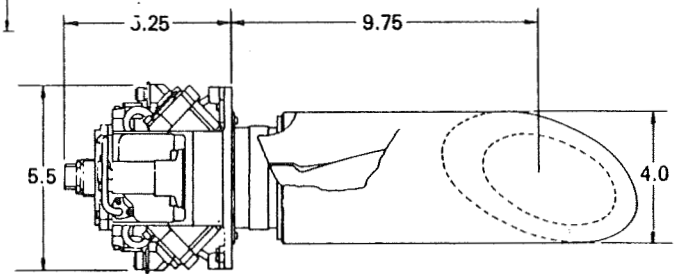
PURPOSE: PROVIDE PROPULSIVE THRUST FOR ORBIT STABILIZATION AND ORIENTATION MANEUVERS

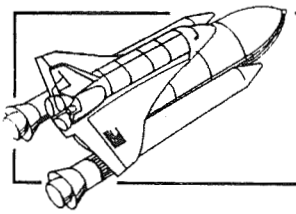
SUPPLIER: THE MARQUARDT COMPANY, VAN NUYS, CA.



PRIMARY

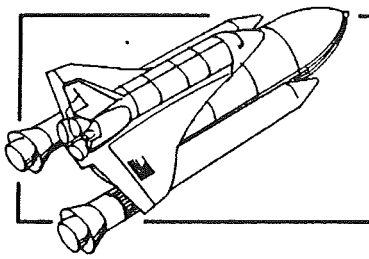
VERNIER



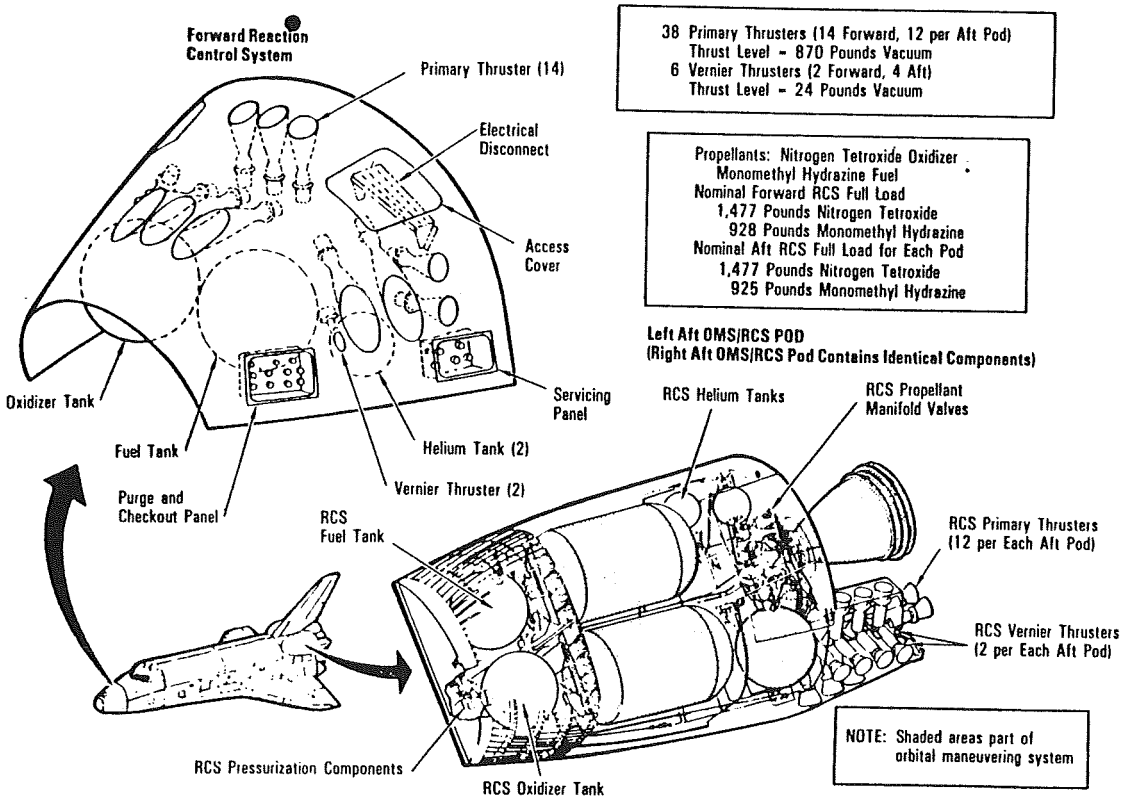


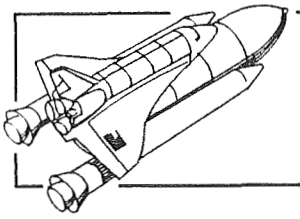
RCS PRIMARY & VERNIER THRUSTER PARAMETERS

	PRIMARY	*VERNIER*
• PROPELLANTS	MMH/N_2O_4	MMH/N_2O_4
• NOMINAL VACUUM THRUST	870 LBS	24 LBS
• CHAMBER PRESSURE	152 PSIA	110 PSIA
• MIXTURE RATIO	1.6	1.65
• SPECIFIC IMPULSE	280 SEC (22:1 AREA RATIO)	265 SEC
• INLET PRESSURE	238 PSIA	246 PSIA
• RATIO (A_e/A_t)	22:1 TO 30:1	20.7:1
• LIFE		
MISSIONS	100	CHAMBER LIMITED
CYCLES	20,000	330,000
TOTAL FIRING DURATION	12,800 SEC	125,000
• WEIGHT	16 LBS	9.4 LBS
• CONSTRUCTION	COLUMBIUM/TITANIUM	COLUMBIUM/TITANIUM



ORBITER OMS & REACTION CONTROL SYSTEM





SPACE SHUTTLE PROPULSION ISSUES

RSRM

- IGNITER SEAL ANOMALIES
- CASE STIFFENER SEGMENT ATTRITION
- IMPROVED O-RING MATERIAL
- ASBESTOS-FREE INSULATION
- FORWARD SEGMENT GRAIN REDESIGN

SRB

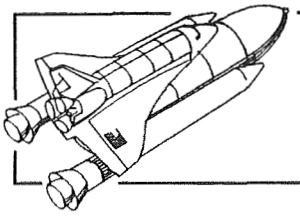
- AFT SKIRT FACTOR OF SAFETY
- OBSOLESCENCE OF ELECTRONIC COMPONENTS
- RECOVERY SYSTEM MARGINS
- DEBRIS CONTAINMENT SYSTEM

SSME

- HIGH PRESSURE TURBOPUMP BEARINGS
- HEAT EXCHANGER
- CONTROLLER OBSOLESCENCE
- UNINSPECTABLE WELDS

RCS THRUSTERS

- COMBUSTION INSTABILITY
- CONTAMINATION



PROPULSION SYSTEM
IMPROVEMENTS IN WORK

RSRM

- IGNITER-TO-CASE JOINT REDESIGN

SRB

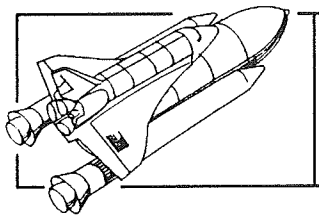
- ENHANCED MULTIPLEXER/DEMULTIPLEXER
- DEBRIS CONTAINMENT SYSTEM FRANGIBLE LINK
- MAIN PARACHUTE RIPSTOP
- HDP/AFT SKIRT BIAS

SSME

- PHASE II + POWERHEAD
- HPOTP/HPFTP LIFE IMPROVEMENTS
- ALTERNATE TURBOPUMP DEVELOPMENT
- BLOCK II CONTROLLER
- SINGLE COIL HEAT EXCHANGER

ORBITER

- IMPROVED AUXILIARY POWER UNIT
- IMPROVED AUXILIARY POWER UNIT CONTROLLER
- IMPROVED MULTIPLEXER/DEMULTIPLEXER



ASA PROGRAM DEFINITION

OBJECTIVE: EXTEND THE LIFE OF THE SPACE SHUTTLE PROGRAM TO THE YEAR 2020

BENEFITS: PLANS FOR OBSOLESENCE, IMPLEMENTS CURRENT TECHNOLOGY

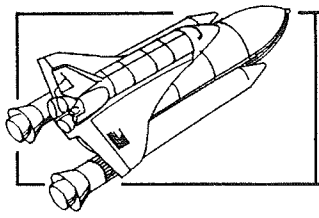
INCREASES SAFETY MARGINS

INCREASES MISSION SUCCESS PROBABILITY

MAINTAINS A HIGH LEVEL OF TECHNICAL EXCELLENCE

IMPROVES VEHICLE TURNAROUND AND OPERATIONS COSTS

DEVELOPS AND QUALIFIES ALTERNATE SOURCES



ASA PROGRAM
SELECTION METHODOLOGY

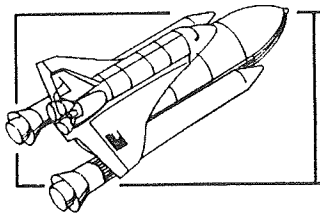
PROBLEM AREAS IDENTIFIED

CANDIDATES SUBMITTED

VIABLE CANDIDATES CATEGORIZED

FEASIBILITY STUDIES BEGUN ON SOME CANDIDATES

CANDIDATES BEING PRIORITIZED

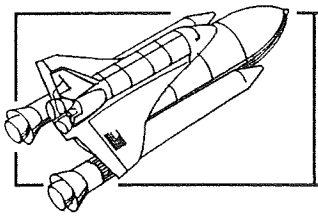


ASA PROGRAM
PRIORITIES

PROGRAM PRIORITIES ESTABLISHED

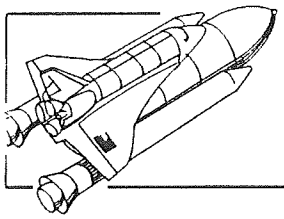
PRIMARY: ASSURANCE OF SYSTEM SUPPORTABILITY AND SAFETY MARGIN IMPROVEMENT

SECONDARY: IMPROVEMENTS IN SYSTEM RELIABILITY, ECONOMY AND PERFORMANCE



ASA PROGRAM
CANDIDATES

TITLE	PROJECT
COCKPIT DISPLAYS AND CONTROLS	ORBITER
EPD&C SUBSYSTEM REDESIGN	ORBITER
CONTROL SYSTEM REDESIGN	SRB
INTEGRATED COMMUNICATIONS	ORBITER
AFT SKIRT REDESIGN	SRB
INTEGRATED OMS/RCS	ORBITER
REDESIGNED STIFFENER RING	RSRM
IGNITER JOINT IMPROVEMENT	RSRM
INTEGRATED NAVIGATION SYSTEM	ORBITER
PROCESS CHEMICALS	SSME
LONG-LIFE FUEL CELLS	ORBITER
COMPOSITE STRUCTURES	SRB
POWERHEAD UPGRADE	SSME
ENHANCED CONTROLLER	SSME
LIGHTWEIGHT STRUCTURES	ORIBTER
INTEGRATED THERMAL CONTROL	ORBITER
FWD SEGMENT MANDREL REDESIGN	RSRM
ALUMINUM LITHIUM ALLOYS	ET
ELECTROMECHANICAL ACTUATORS	ORB/SSME



ASA PROGRAM
CATEGORIES

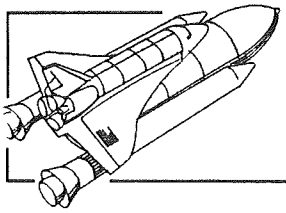
A. HIGHEST PRIORITY

NEAR TERM SUPPORTABILITY ISSUES
SAFETY MARGIN INCREASES

B. HIGH PRIORITY-SYSTEMS IMPROVEMENTS WITH IMPLEMENTATION OPPORTUNITIES

C. OTHER IMPROVEMENTS WITH INDEFINITE SCHEDULE DRIVERS

D. IMPROVEMENTS WITH NO SCHEDULE DRIVER AND/OR HIGH PROGRAM RISK



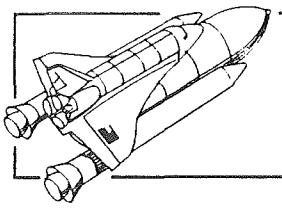
ASA PROGRAM
PROPULSION PROGRAM CANDIDATES

SRB CONTROL SYSTEM REDESIGN

SSME ADVANCED FABRICATION

AFT SKIRT REDESIGN

INTEGRATED OMS/RCS



ASA PROGRAM
SRB CONTROL SYSTEM REDSIGN

DESCRIPTION:

REPLACE OBSOLETE ELECTRONIC CONTROL SYSTEMS (FORWARD & AFT IEA'S) WITH SINGLE INTEGRADED MICROPROCESSOR SYSTEM

ADD SOLID PROPELLANT APU GAS GENERATOR TO REPLACE HYDRAZINE SYSTEM

ADD NEW LASER INITIATED ORDNANCE TO REPLACE CURRENT SYSTEM

BENEFITS:

SMART INTEGRATED ELECTRONICS ASSEMBLIES (IEA) AND RANGE SAFETY DISTRIBUTER (RSD) CONTROLLERS AND LASER ORDNANCE CONTROLS ELIMINATES COMPONENTS, FAILURE MODES AND REDUCES COSTS

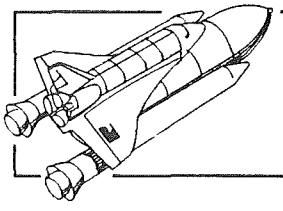
EXTERNALLY PROGRAMMABLE MICROPROCESSOR SYSTEM

HIGHER LAUNCH PROBABILITY FROM REDUCED WING LOADS DUE TO ELIMINATION OF AFT IEA PROTRUBERANCE

FIBER OPTIC DATA BUSES FOR BETTER COMMUNICATIONS

ELIMINATE ORDNANCE SYSTEM EMI CONCERNS WITH FIBER OPTIC LINES

ELIMINATE HYDRAZINE CONCERNS



ASA PROGRAM
SRB AFT SKIRT REDESIGN

DESCRIPTION:

NEW AFT SKIRT, DESIGN TO:

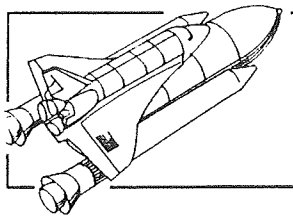
- INCREASE STRUCTURAL FACTOR OF SAFETY (1.28 TO 1.4)
- ENHANCE HOLDDOWN MECHANISM
- ADD INTEGRAL STIFFENER RINGS TO MINIMIZE WATER IMPACT DAMAGE

BENEFITS:

SAFETY MARGIN ENHANCEMENT

ELIMINATE STUD HANGUP AND LAUNCH LOADS

REDUCTION IN WATER IMPACT DAMAGE



ASA PROGRAM
SSME ADVANCED FABRICATION

DESCRIPTION:

MAJOR REDESIGNS EMPLOYING ADVANCED FABRICATION AND CASTING TECHNIQUES TO RESOLVE MAJOR ISSUES:

- FINE GRAINED INVESTMENT CASTINGS
- VACUUM PLASMA SPRAY FOR MAIN COMBUSTION CHAMBER

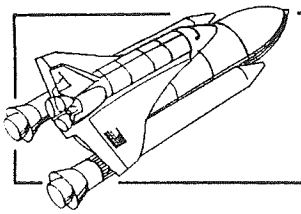
BENEFITS:

IMPROVE THE INSPECTABILITY OF CRITICAL WELDS

ELIMINATE 3000 UNINSPECTABLE WELDS

REDUCE FABRICATION COSTS OF MAJOR COMPONENTS

INCREASE DESIGN PERFORMANCE MARGIN



ASA PROGRAM
INTEGRATED OMS/RCS

DESCRIPTION

REDESIGN SEPARATE OMS/RCS SYSTEMS INTO ONE INTEGRATED SYSTEM
- ELIMINATE RCS TANKS/PRESSURIZATION SYSTEM
- ALLOW OMS TANK PLUS ENTRY SUMP USE FOR BOTH OMS AND RCS PROPELLANT
- IMPROVE ABORT DUMP CAPABILITY
- ALLOW LANDING WITH INCREASED RESIDUAL PROPELLANT
- INCREASE CHECKOUT/MAINTENANCE CAPABILITY WITH POD ON ORBITER

BENEFITS

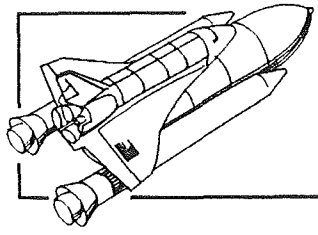
IMPROVE SAFETY MARGIN

REDUCE COST

SIMPLIFIED MISSION PLANNING

350 LB DRY WEIGHT REDUCTION

RETAIN CONTRACTOR/SUBCONTRACTOR DESIGN/PRODUCTION SKILLS



ASA PROGRAM
SUMMARY

THE SHUTTLE LIFE CYCLE CAN BE EXTENDED FROM 20 TO 40 YEARS

SIGNIFICANT BUDGET SAVINGS CAN BE REALIZED OVER A NEW SHUTTLE II

SUBSYSTEM MANDATORY UPGRADES FOR OBSOLESCENCE, SAFETY MARGIN, AND PERFORMANCE IS REQUIRED TO EXTEND THE SHUTTLE LIFE

UPGRADE PROGRAMS WILL HAVE A DEDICATED MANAGEMENT SYSTEM

UPGRADES WILL BE TIMED FOR EFFICIENT IMPLEMENTATION

UPPER STAGES/PROPULSION

87

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5P

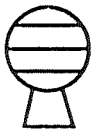
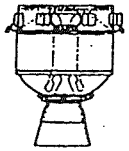
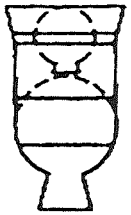
N91-28201

CURRENT SYSTEMS

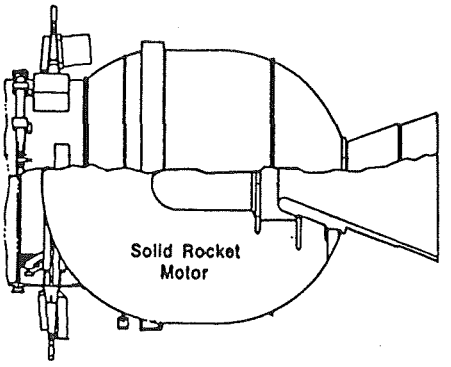
UPPER STAGES

CHARLES R. GUNN
NASA
OFFICE OF SPACE FLIGHT
JUNE 26, 1990

UNITED STATES ORBITAL TRANSFER VEHICLES

CHARACTERISTICS			PAM-D	PAM-DII	TOS	IUS
STAGE:	MANUFACTURER		MDAC	MDAC	MMC	BAC
	LENGTH	(FT)	6.75	6.5	10.0	16.4
	DIAMETER	(FT)	4.0	5.3	11.3	9.5
ENGINE:	MANUFACTURER		THIOKOL	THIOKOL	CSD	CSD
	TYPE		(STAR 48)	ISTP	SRM-1	
	NUMBER		1	1	1	SRM-1 SRM-2
	FUEL		SOLID	SOLID	SOLID	SOLID
	COMPOSITION		TP-H-3340	--	HTPB	HTPB
TOTAL THRUST		(LB)	14,500	17,600	45,000	45,000 18,250
SPECIFIC IMPULSE		(SEC)	205.6	--	294	292.9 300.9
BURN TIME		(SEC)	85.0	121	150	153.0 104.8
STAGE:	PAD WEIGHT	(LB)	4,616	7,690	23,700	32,537
	IMPULSE PROPELLANT WEIGHT	(LB)	4,400	7,150	21.400	21,403 1,050
	BURNOUT WEIGHT	(LB)	418	540	2,390	2,553 2,523
	AIRBORNE SUPPORT EQUIP. WT	(LB)	2,505	3,525	3,200	7,377
PAYLOAD:	TO GEO ONE-WAY STAGE	(LB)	1,400	2,100	6,600	5,090
	TO GEO TRANSFER ORBIT (GTO)	(LB)	2,750	4,160	13,000	~17,000
ILLUSTRATION:			 PAD-D	 PAM-DII	 TOS	 IUS (TWO STAGE)
SCHEDULE:	START DATE		1976	1980	1983	1978
	OPERATIONAL DATE		1982	1985	1986	1982
TYPE OF DEVELOPMENT			COMMERCIAL	COMMERCIAL	COMMERCIAL	U.S. GOV'T
SPONSOR			MDAC	MDAC	OSC	USAF

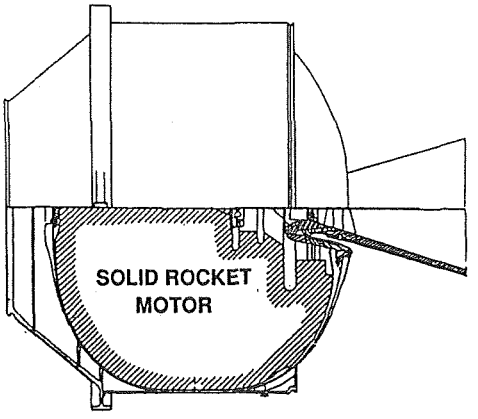
PAM-D



● COMPATIBILITY:	DELTA II AND SPACE SHUTTLE
● PERFORMANCE CAPABILITY:	2,700 POUNDS GEOSYNCH TRANSFER
● FLIGHT RECORD:	95% (40 / 42)
● COST:	$6 to 7 MILLION DOLLARS

* 160 x 19,323 Nmi (296 x 35,786 Km)

PAM-DII



● COMPATIBILITY:	TITAN III AND SPACE SHUTTLE
● PERFORMANCE CAPABILITY:	4,000 POUNDS GEOSYNCH TRANSFER*
● FLIGHT RECORD:	100% (2 / 2)
● COST:	$10 to 12 MILLION DOLLARS

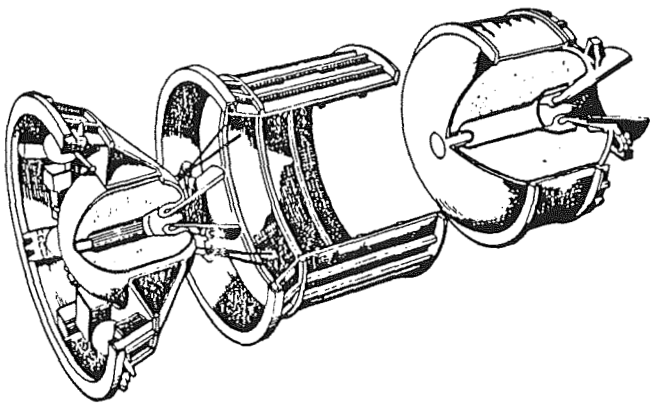
* 160 x 19,323 Nmi (296 x 35,786 Km)

TOS



• COMPATIBILITY:	TITAN III AND SPACE SHUTTLE
• PERFORMANCE CAPABILITY:	5,000 to 13,400 POUNDS GEOSYNCH TRANSFER
• FLIGHT RECORD:	
• COST:	$35 to 45 MILLION DOLLARS

IUS

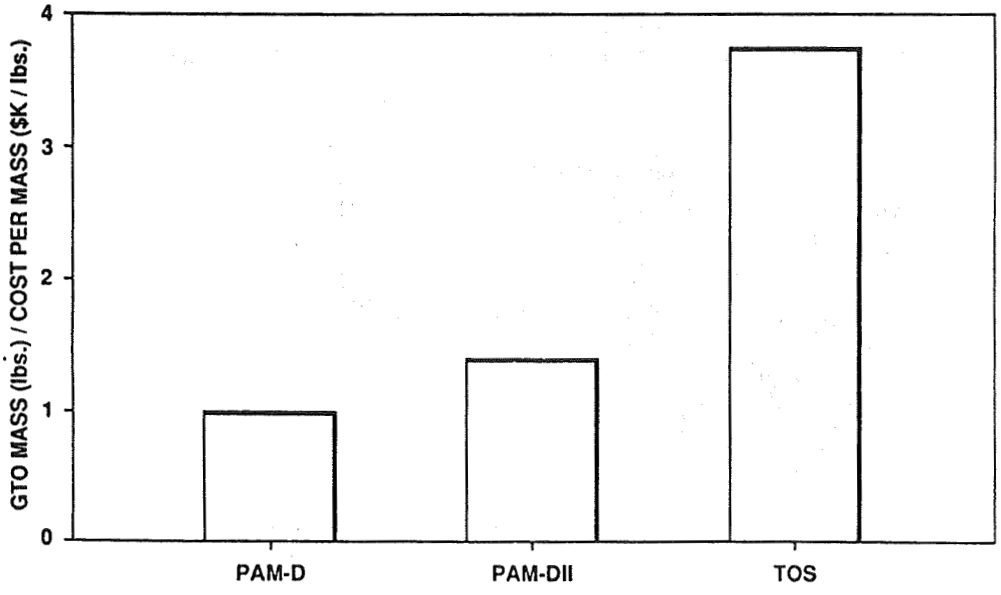


• COMPATIBILITY:	TITAN IV AND SPACE SHUTTLE
• PERFORMANCE CAPABILITY:	5,000 POUNDS IN GEOSYNCH
• FLIGHT RECORD:	86% (6 / 7)
• COST:	$60 to 70 MILLION DOLLARS

POTENTIAL
NASA UPPER STAGE MISSIONS

- LUNAR OBSERVER - 1996
- MARS OBSERVER FOLLOW-ON - 1996
- ADVANCED TDRS (SERIES OF 9) - 1997

U.S. ORBITAL TRANSFER VEHICLES
COST EFFECTIVENESS



88

540704
25P

N91-28202

UNITED TECHNOLOGIES PRATT&WHITNEY

CRYOGENIC UPPER STAGE PROPULSION

RL10 and Derivative Engines

Presented to:
Space Transportation Propulsion
Technology Symposium
Pennsylvania State University

June 1990

Presented by:
James R. Brown
Manager, Upper Stage Programs

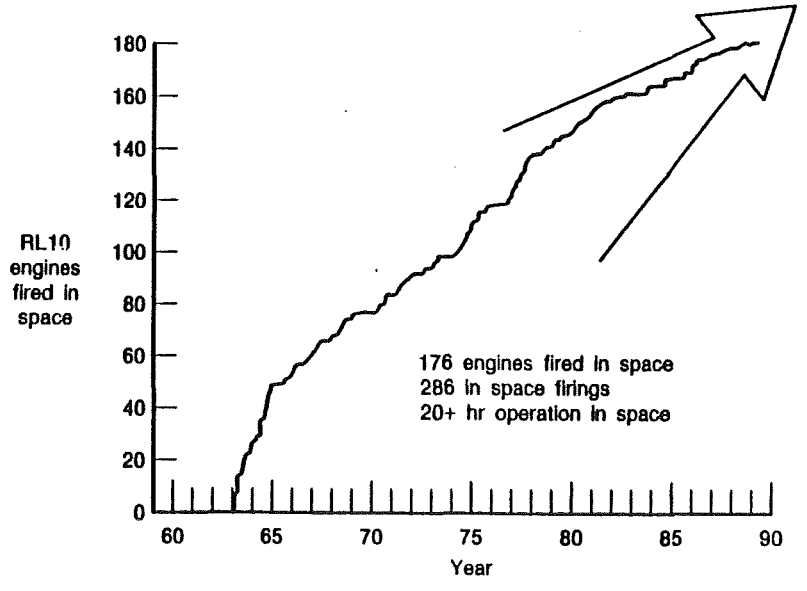
SPACE PROPULSION AND SYSTEMS
P. O. Box 109600
West Palm Beach, Florida 33410-9600

TOPICS

- Engine model history
- RL10 demonstrated capabilities
- RL10 derivative potential for SEI
- Summary

RL10 LIQUID HYDROGEN ROCKET ENGINE

- Perfect flight record - 100% reliable



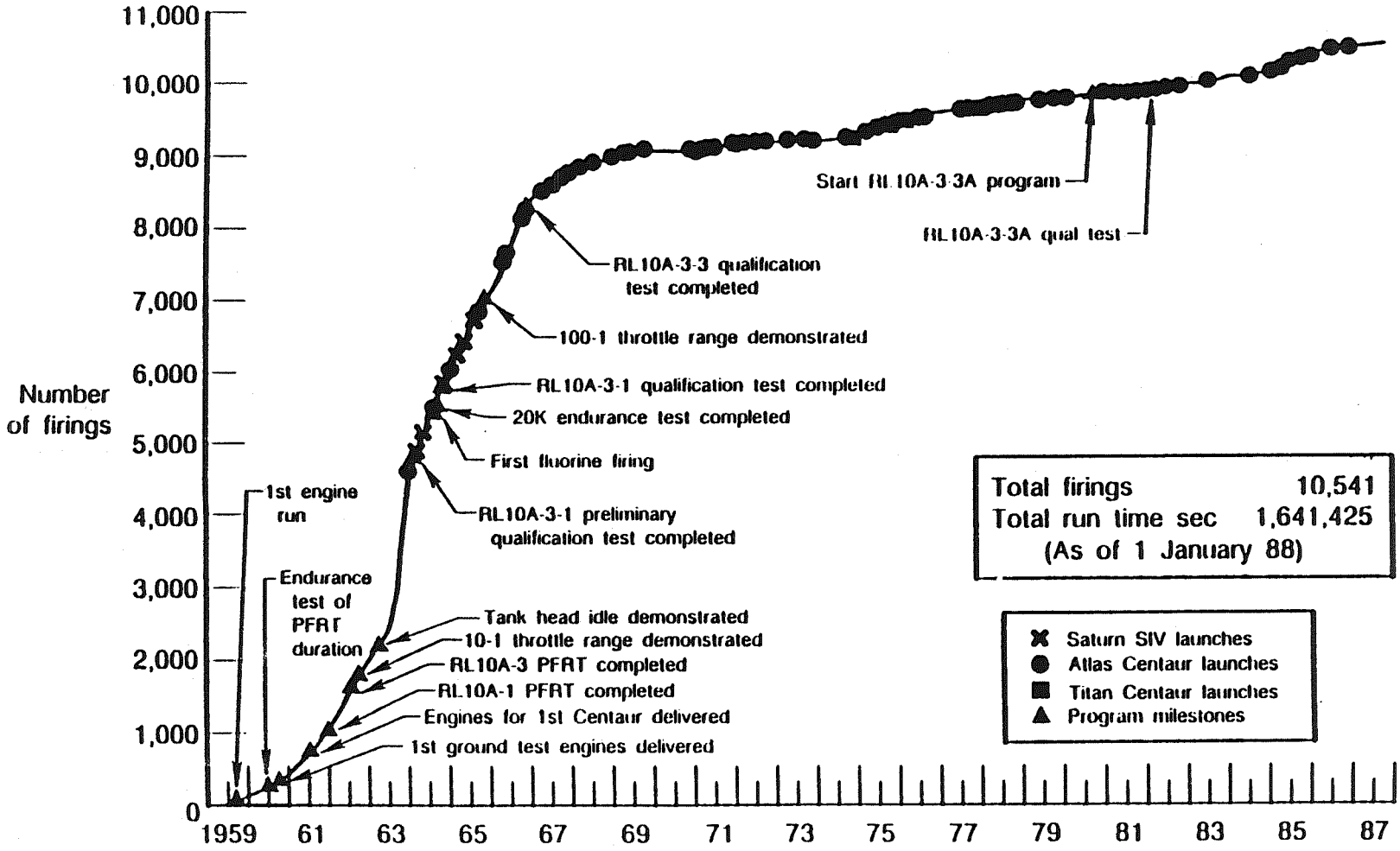
RL10A-3-3A ENGINE

Vacuum thrust, lb	16,500
Specific impulse, sec	444.4
Weight, lb	305
Mixture ratio	5:1
Chamber pressure, psia	475
Area ratio	61:1
Qual life, firings/hr	20/1.25

RL10 EVOLUTION

Model no.	A-1	A-3	A-3-1	A-3-3	A-3-3A	A-4
Vac thrust, lb	15,000	15,000	15,000	15,000	16,500	20,800
Chamber pressure, psia	300	300	300	395	475	578
Thrust/weight	50	50	50	50	54	57
Expansion ratio	40:1	40:1	40:1	57:1	61:1	84:1
I_{sp}, sec at 5.0 O/F (5.5)	424	429	433	442.4	444.4	(449.0)
Flight certification date	11/61	6/62	9/64	10/66	11/81	12/90

RL10 DEVELOPMENT HISTORY



RL10 EXPERIENCE

Demonstrated capability

- High ratio nozzle
- Durability
- Low thrust/throttling
- Higher thrust
- High mixture ratio
- Alternate propellants

HIGH AREA RATIO NOZZLE

- Extensive testing with 84 area ratio extensions
- Boilerplate 205 area ratio tested
- High area ratio contour primary nozzle tested
- Extending / retracting system under development

CONCEPT VERIFICATION 84 AREA RATIO NOZZLE TESTING

RL10 engine installed in E-6 test stand	Carbon/carbon nozzle during engine run
Carbon/carbon nozzles (2)	9,233 sec (69 firings)
Columbium nozzles (4)	5,392 sec (26 firings)
Total	14,625 sec (95 firings)

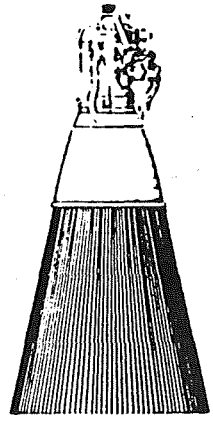
RL10 EXPERIENCE

205 area ratio boilerplate nozzle results

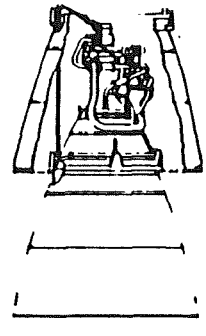
RL10A-3-3



RL10A-3-3 with ε = 205 Nozzle Extension



Short RL10 Derivative 11B ε = 205

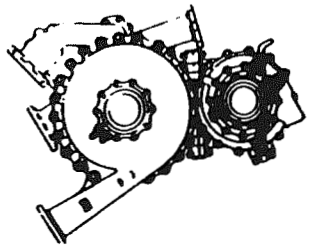


Δ Specific Impulse (ε = 57 to ε = 205)	20.9 sec @ O/F = 5.0 20.4 sec @ O/F = 6.0

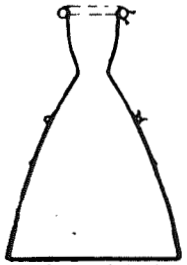
RL10 EXPERIENCE

Durability

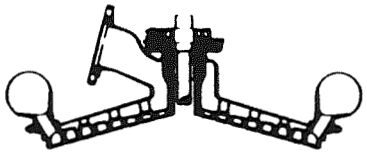
- Life potential greater than 5.0 hours, 200 firings at 15K thrust



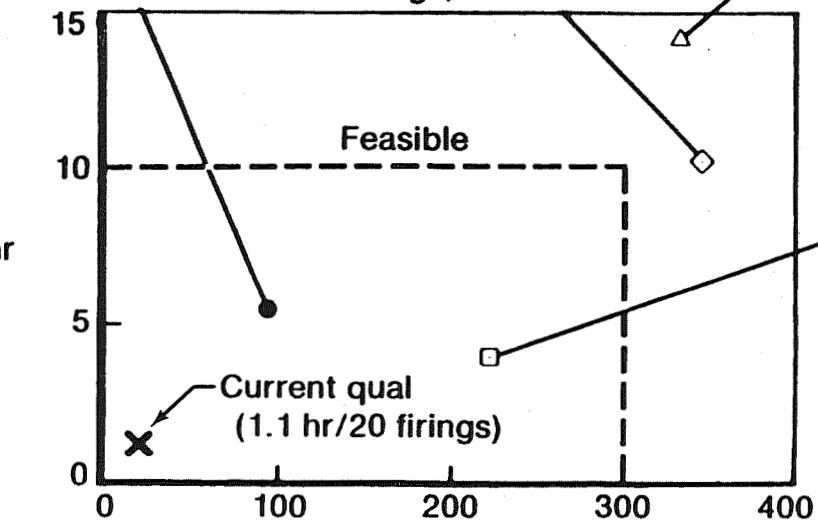
Turbopump assembly =
89 firings, 5.56 hr



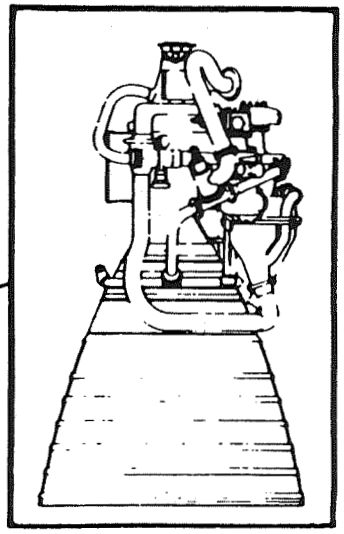
Thrust chamber high time =
345 firings, 10.5 hr



Injector high time =
337 firings, 14.4 hr



Increased life capability



Engine high time for one build = 223 firings, 4.0 hr
86 engines have exceeded 3600 sec in a single build

RL10 EXPERIENCE

Durability

- Rubbing carbon seal life limit on turbo machinery
 - Function of velocity - proportional to speed
 - Gears may be limit above 25K thrust
- Chamber low cycle fatigue limit on number of firings
 - Most severe strain during start transient

RL10 EXPERIENCE

Low Thrust

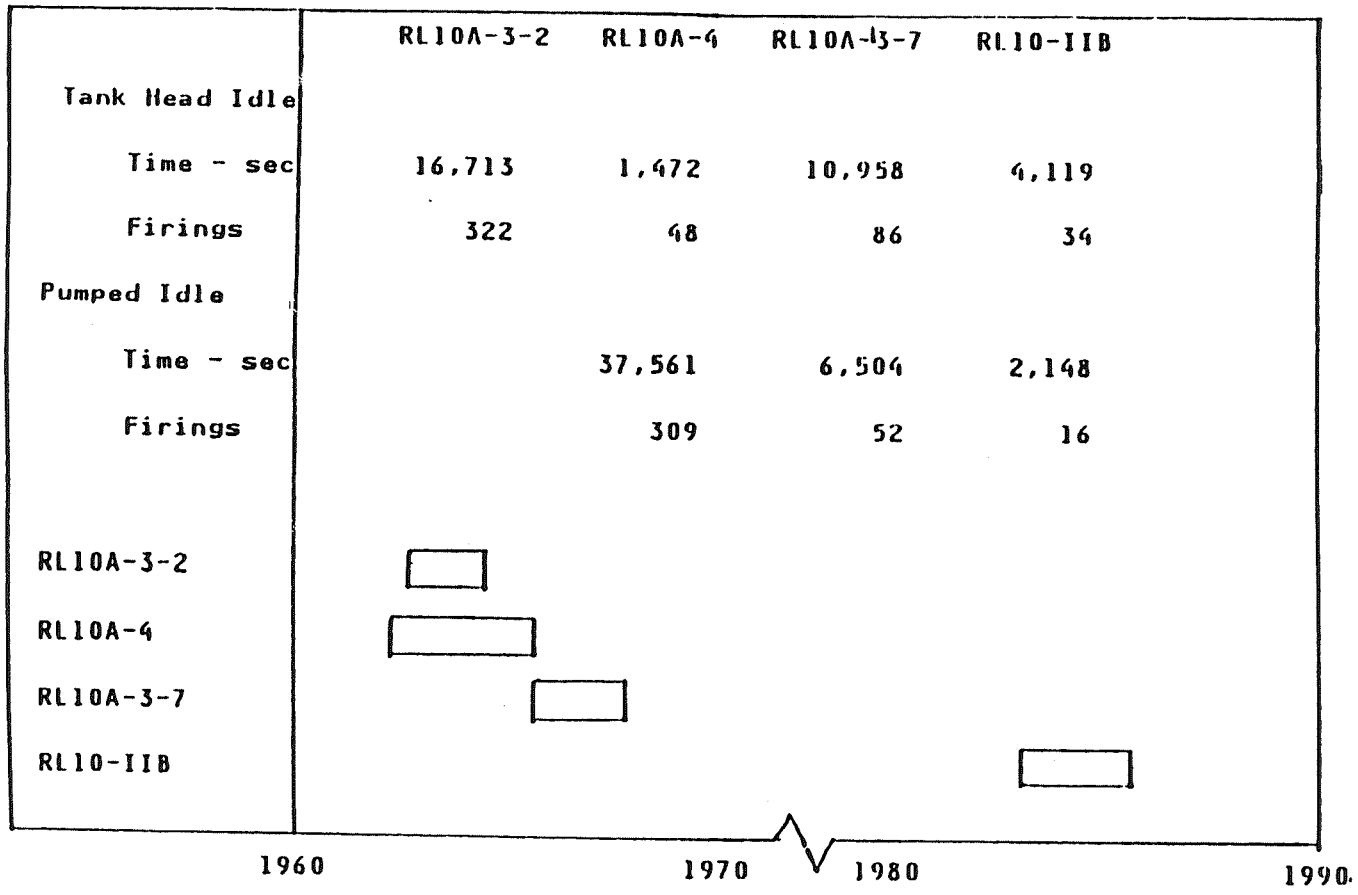
1960's Testing
* Full Throttling
* Complex Controls
* High Loss Injector

1980's Testing
* Stepped Thrust Levels
* Simple Controls
* Gox Heat Exchanger

RL10 EXPERIENCE

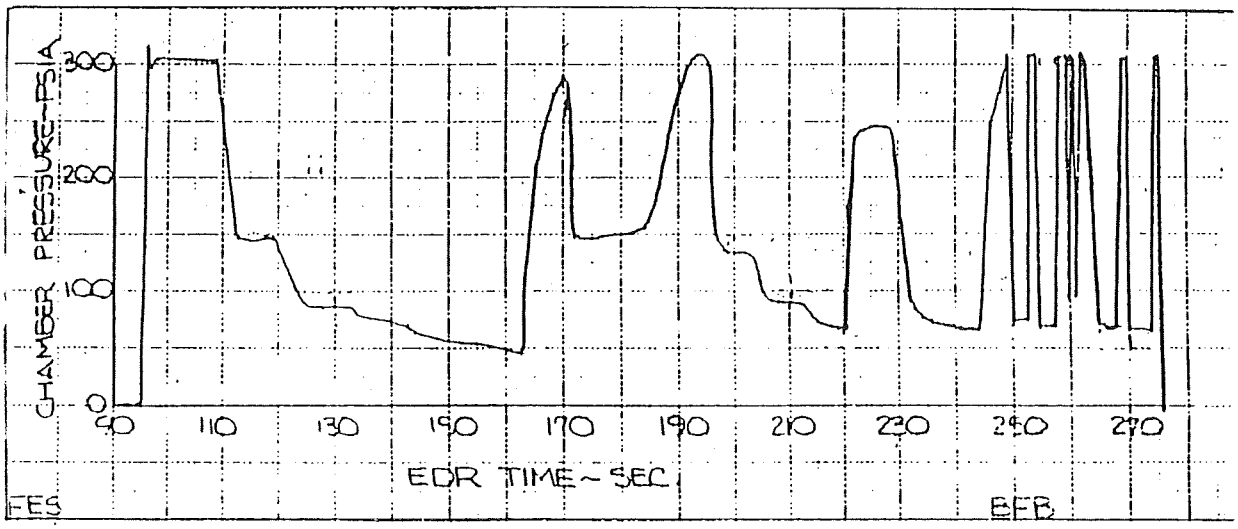
RL10 low thrust summary

	RL10A-3-2	RL10A-4	RL10A-3-7	RL10-IIB
Tank Head Idle				
Time - sec	16,713	1,472	10,958	4,119
Firings	322	48	86	34
Pumped Idle				
Time - sec		37,561	6,504	2,148
Firings		309	52	16



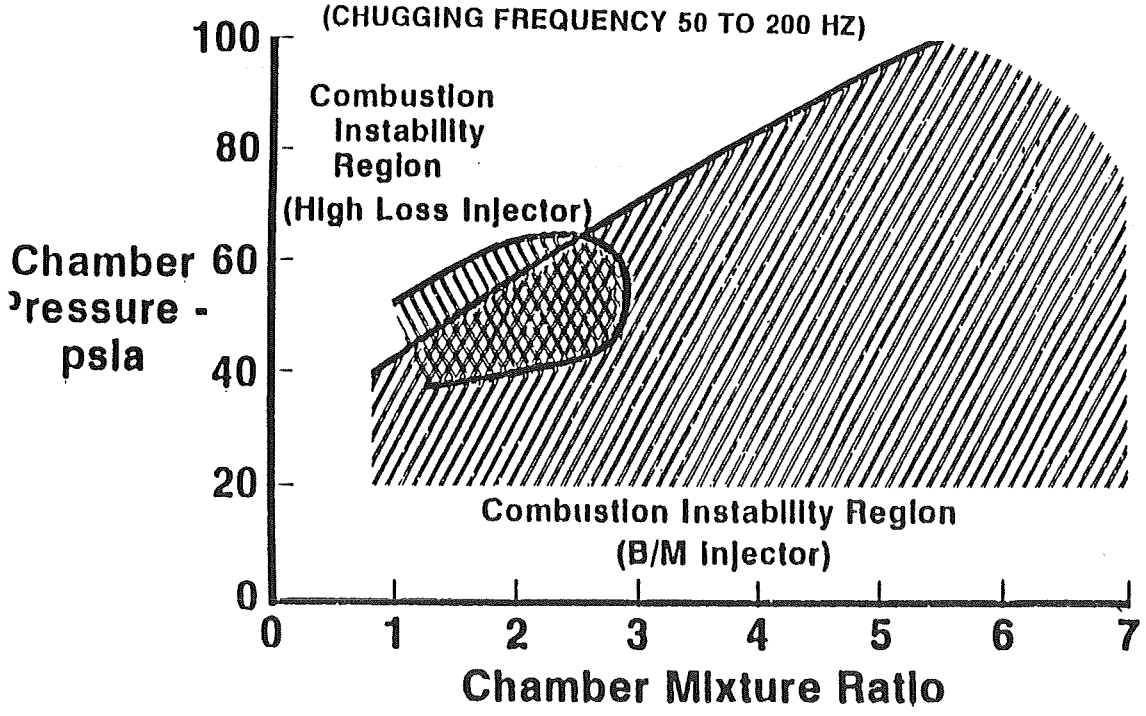
RL10 EXPERIENCE

Typical RL10A-4 throttling test



RL10 EXPERIENCE

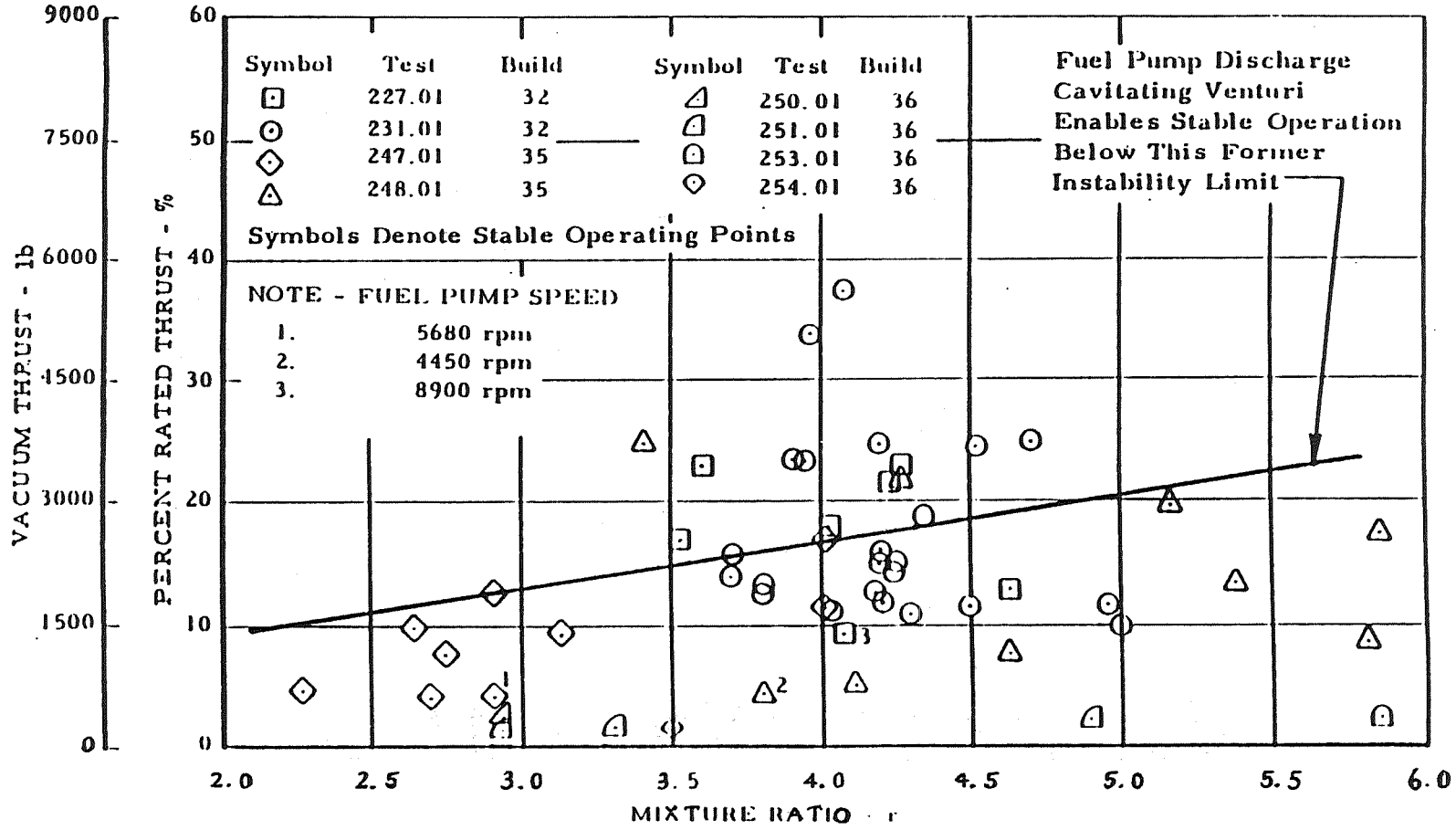
RL10 combustor instability characteristics



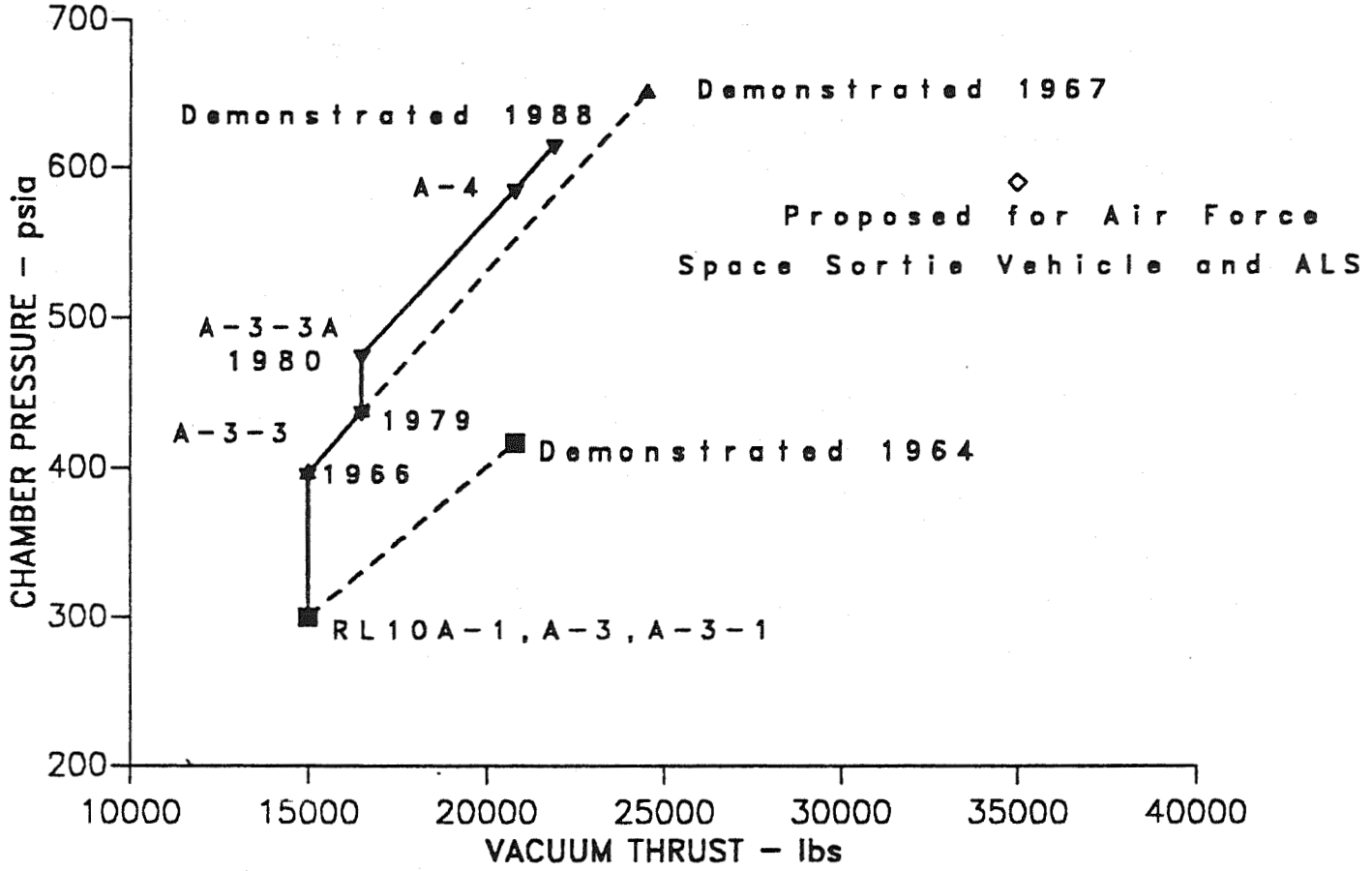
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RL10 EXPERIENCE

Stable throttling to less than 2%



RL10 UPRATING HISTORY



RL10 EXPERIENCE

High mixture ratio summary

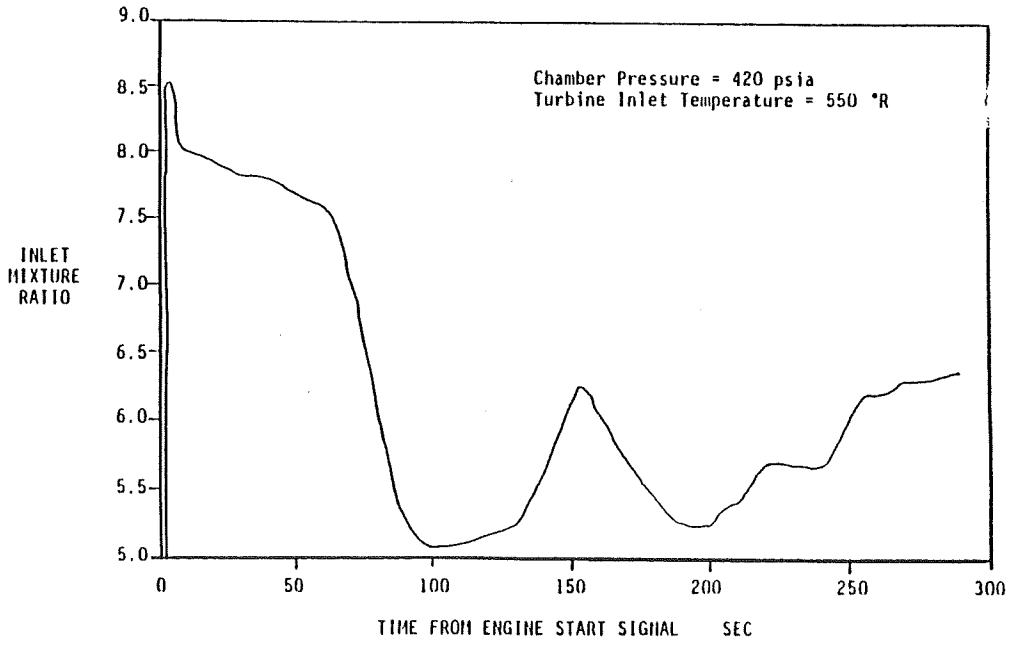
- 250 sec greater than 7.0 O/F - - 7 engines - -
- One run up to 13.5 O/F

 - - 5.0 to 13.5 in 35 sec
- One run over 7.5 O/F more than 60 sec

NO ENGINE DAMAGE FROM HIGH O/F OPERATION!!

RL10 EXPERIENCE

Stoichiometric mixture ratio demonstrated



RL10 EXPERIENCE

Alternate Propellants

FLOX/CH4 9 Tests 120 Sec 11K Thrust

O2/C3H8;;23 Tests 106 Sec 14K Thrust

F2/H2 29 Tests 1757 Sec 21K Thrust

RL10 CAPABILITY

Space initiative space propulsion requirements

- Throttling
- High performance
- Reusable
- Space based
- Man rated

RL10 CAPABILITY

Throttling

Throttling engine testing 1963 through 1965 (under MSFC contract) most applicable

- 309 Tests
- 37,561 Seconds
- Demonstrated stable, continuous throttling down to 2% thrust
- Solved oxidizer chugging stability and fuel system stability problems

RL10 CAPABILITY

Specific modifications for continuous throttling

- Control valves
 - Fuel control (turbine bypass)
 - Oxidizer control
 - Cavitating venturi
- Scheduler (controller)
- Idler gear ratio (from 2.5 to 2.13)
- High loss injector

RL10 CAPABILITY

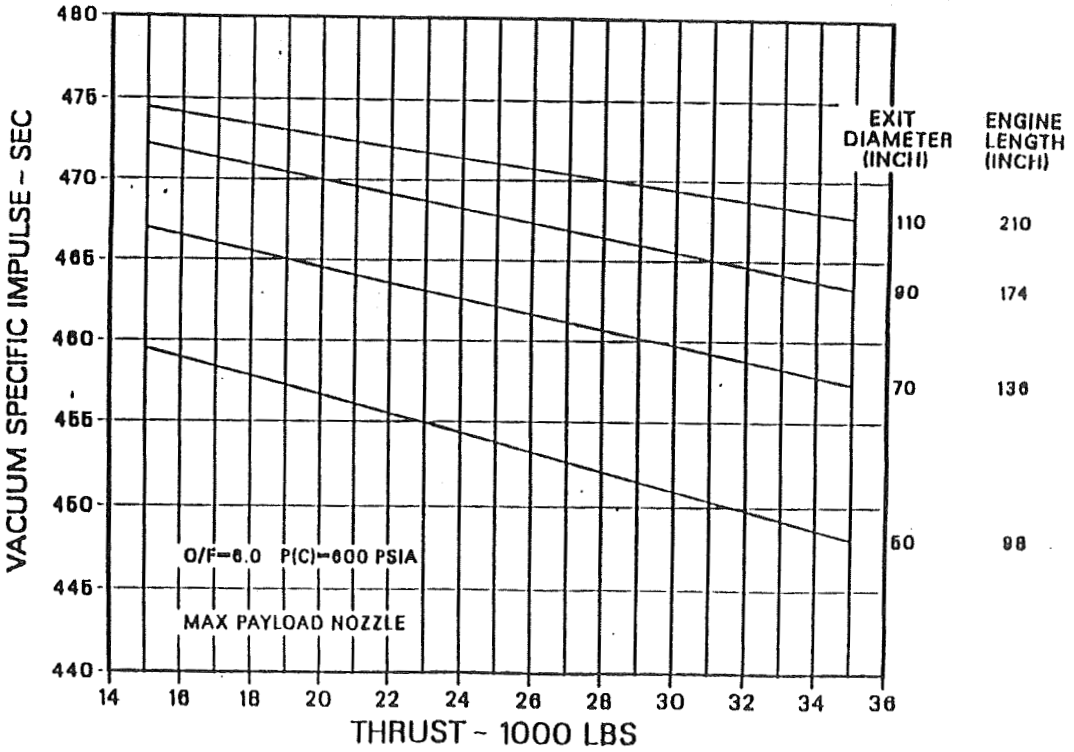
High performance

- High area ratio nozzle
 - Extensive testing 84 area ratio extensions
 - Boilerplate 205 area ratio tested
 - Extending/retracting system under development

- Impulse potential > 470 sec

- Available impulse dependent on envelope constraints

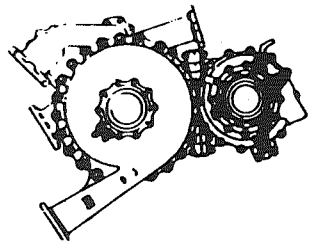
EFFECT OF THRUST LEVEL ON PERFORMANCE



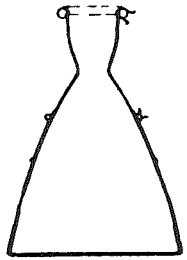
RL10 CAPABILITY

Reusable

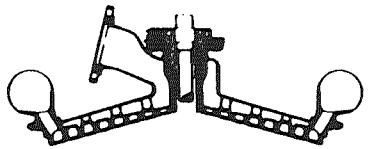
- Life potenital greater than 5.0 hours, 200 firings



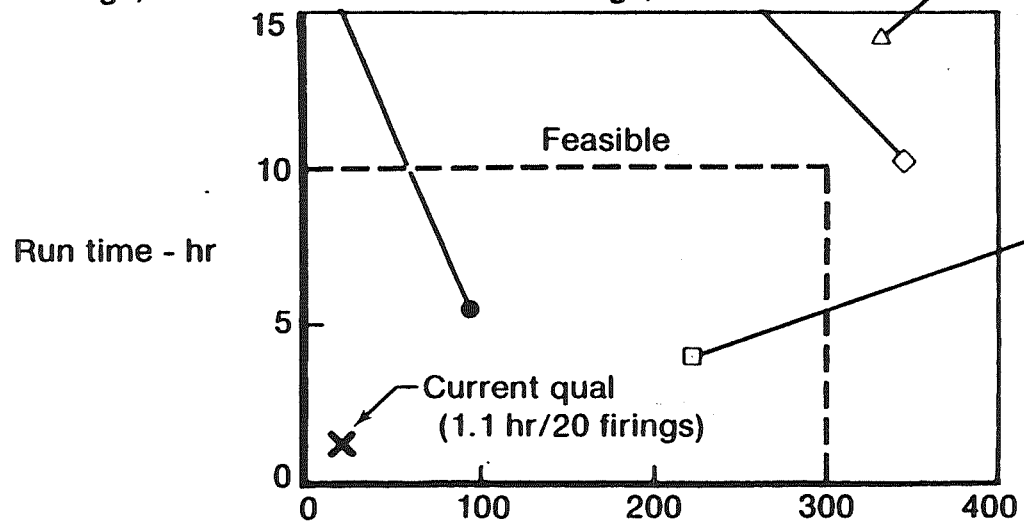
Turbopump assembly =
89 firings, 5.56 hr



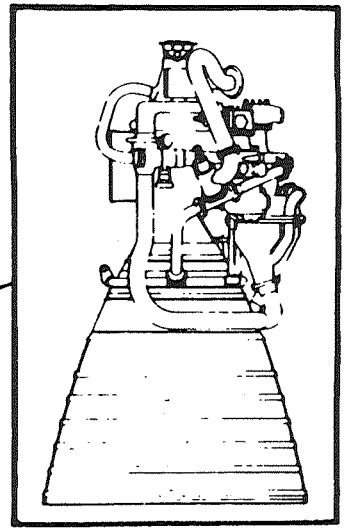
Thrust chamber high time =
345 firings, 10.5 hr



Injector high time =
337 firings, 14.4 hr



Increased life capability



Engine high time for one build = 223 firings, 4.0 hr
86 engines have exceeded 3600 sec in a single build

MAN-RATING

- High reliability
- Noncatastrophic failure modes
- Redundant components and/or engine out

RL10 CAPABILITY

Man-rated

- RL10 has high demonstrated reliability
 (0.9984 @ 90% confidence level)
- RL10 has benign failure modes
 (Current base model - RL10A-3-3 has > 3,800 tests
 over 25 years with no catastrophic engine failures)
- With engine-out capability, RL10 propulsion system would have very high reliability and safety

SPACE-BASING

- Minimal maintenance needed
- Health monitoring provided
- Easy engine change-out capable
- Long space exposure compatible

RL10 CAPABILITY

Space based

- Only minor maintenance practical in space environment
- Removal of entire engine most likely solution for problem
- RL10 could be modified to facilitate engine changeout
- RL10 well characterized for health diagnostic purposes
- RL10 has demonstrated reusability in space (7 firings on single mission)

SUMMARY

THE RL10 IS NOT DESIGNED TO BE

- A high pressure engine
- A small envelope engine
- Inherently redundant

AN RL10 DERIVATIVE IS . . .

- Based on in-depth studies/hardware demonstrations
- Low maximum system pressures
- Low program risk
- Near-term available
- Highly reliable due to its simplicity/low pressure
- Well characterized and understood (large data base)
- Turbo machinery configured for full throttling
- Failure tolerant
- Multiple start capability

RL10 DERIVATIVE
Options

- Deep continuous throttling
- Extended operational life
- High area ratio nozzle
- Higher thrust
- Higher mixture ratio
- H_2 and/or O_2 tank pressurization
- Tank head idle
- Health monitoring
- Quick/smart disconnects

RL10 DERIVATIVE OPTION RANGES

	RL10A-4 (New)	RL10 Derivative B Family	RL10 Derivative C Family
Thrust lbs	20,800	15,000 to 22,000	25,000 to 35,000
Nominal mixture ratio	5.5	4.0 to 12.0	
Area ratio	84	up to 600+	
Thrust positions	Full thrust	Multi position or continuous deep throttling	
TBO, firings	15	60+	
HRS	0.8	2+	
Tank pressurization	H_2	H_2 and/or O_2	
Engine conditioning	Dump	Dump or THI	

RL10 CAPABILITY

Summary

- RL10 reliability record unmatched
- Throttling capability demonstrated
- Durability potential demonstrated
- RL10 has demonstrated many capabilities required for space initiative propulsion

PRESENTATION 1.2.4

SATELLITE/SPACECRAFT PROPULSION

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SATELLITE/SPACECRAFT PROPULSION

Presented at
Space Transportation Propulsion Technology Symposium
Penn State University
26 June 1990

by
Mack W. Dowdy, PhD
Propulsion and Chemical Systems Section
Jet Propulsion Laboratory
California Institute of Technology

TYPES OF SPACECRAFT PROPULSION

SOLID ROCKET MOTORS

- Orbit Insertion Maneuvers

LIQUID ROCKET ENGINES

- Main Propulsion

 Orbit Insertion, Trajectory Control

- Attitude Control Propulsion

 Stationkeeping, S/C Pointing, Orbit Makeup

OTHER TYPES

- Hydrazine Resistojets
- Low-Power Arcjets

SATELLITE/SPACECRAFT MISSIONS

NASA

- Solar System Exploration
- Earth Observation Satellites (EOS)
- Space Exploration Initiative (SEI)

COMMERCIAL

- Communications, Relay

MILITARY

- Observation, Tracking, Relay
- Communications, Global Positioning

PROPULSION TECHNOLOGY DRIVERS

PERFORMANCE

- High Temperature Materials, More Energetic Propellant Combinations, High Expansion Ratio Nozzles

CONTAMINATION CONTROL

- Low Contamination Propellants

LONG LIFE

- Propellant/Materials Compatibility, Leak-Free Components, Health Monitoring and Control

REDUCED WEIGHT

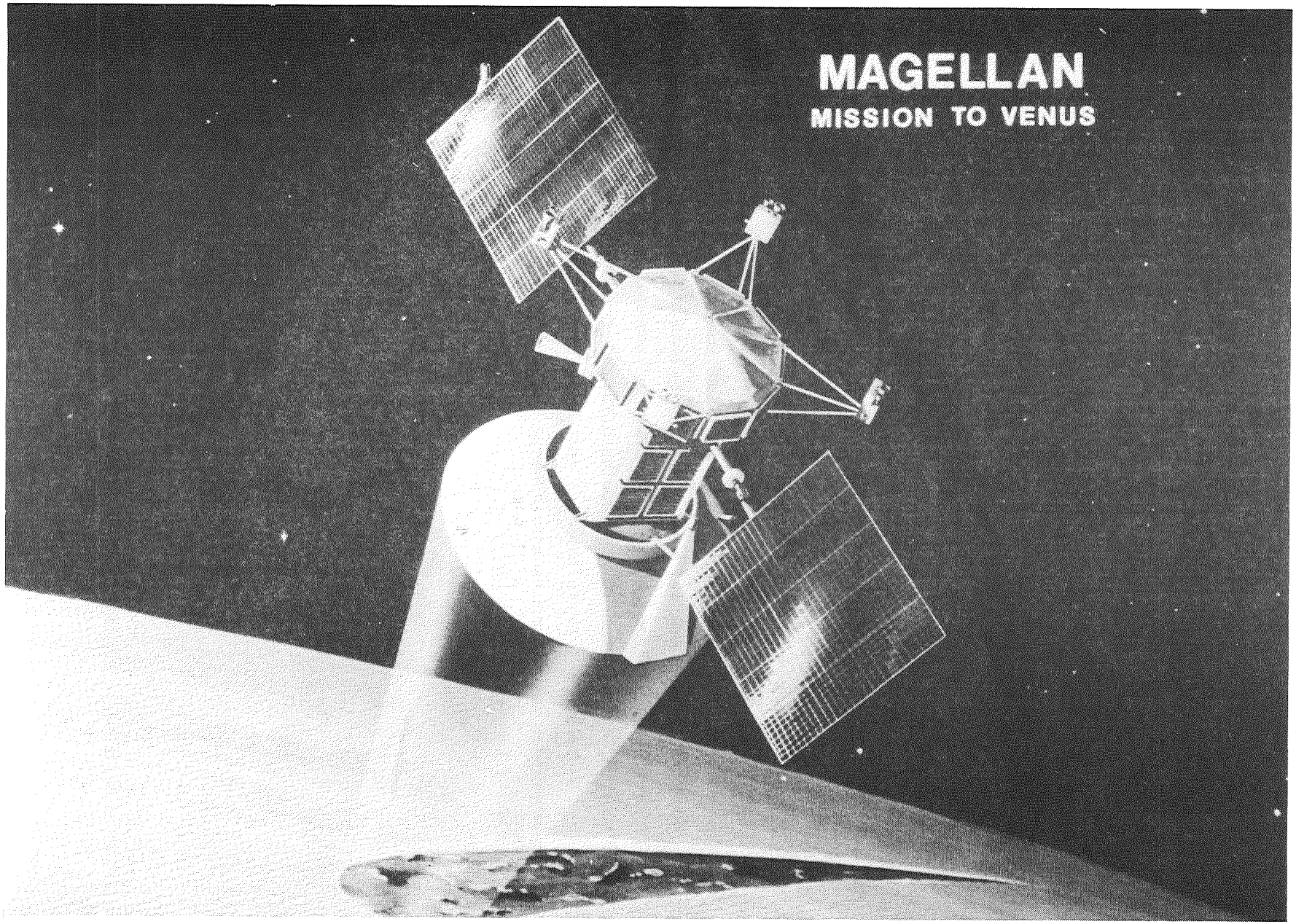
- Engine Performance, Dual Mode Operation, Light Weight Components (Tankage)

RELIABILITY

MAGELLAN PLANETARY SPACECRAFT

The Magellan spacecraft, launched on 4 May 1989, will gather data needed to understand the surface and interior of Venus. Synthetic aperture radar will penetrate the thick Venusian atmosphere to produce photographic-quality images which will inform us of the geological processes that have acted over time to produce the planet's surface.

The propulsion system for the Magellan spacecraft consists of a solid rocket motor (SRM) for the Venus orbit insertion maneuver and a monopropellant hydrazine system for the remaining propulsive maneuvers. The SRM is a Star 48 motor, built by Thiokol, which will be separated from the spacecraft following the orbit insertion maneuver. The monopropellant hydrazine engines are mounted on four similar rocket engine modules (REM). Each REM includes one 22-N engine, three 0.9-N engines and two 445-N engines. The engines are all built by Rocket Research.



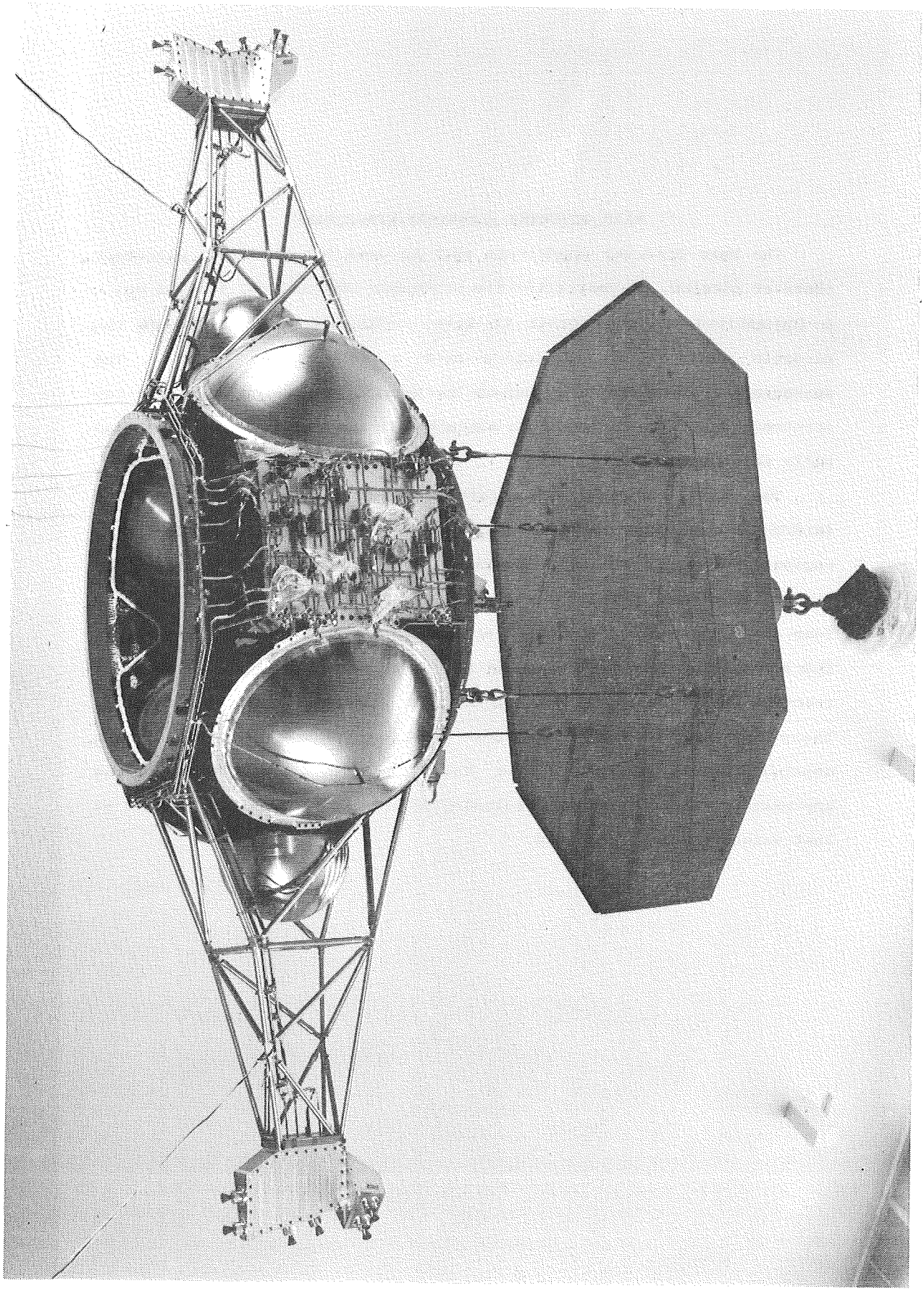
MAGELLAN PLANETARY SPACECRAFT

GALILEO PLANETARY SPACECRAFT

The Galileo spacecraft, launched in October 1989, will conduct the first in-depth exploration of the Jovian system. The spacecraft consists of a probe, which will enter the giant planet's atmosphere, and a sophisticated dual-spinning orbiter, which will study Jupiter, its magnetosphere and its major satellites during a 22-month mission.

Bipropellant engines (NTO/MMH), built by MBB, are used for both main and ACS propulsion functions on the Galileo planetary spacecraft. The spacecraft weighs approximately 2500 kg at the beginning of life, with about 38 percent of that weight being propellants. The main propulsion function is accomplished with one bipropellant engine, producing a thrust of 400-N and operating in both steady-state and pulse mode. The ACS propulsion function is accomplished with twelve bipropellant engines, each producing a thrust of 10-N and operating in the pulse mode. It is expected that some of the 10-N engines could experience 20,000 pulses and a total operating time of seven hours during the Galileo mission.

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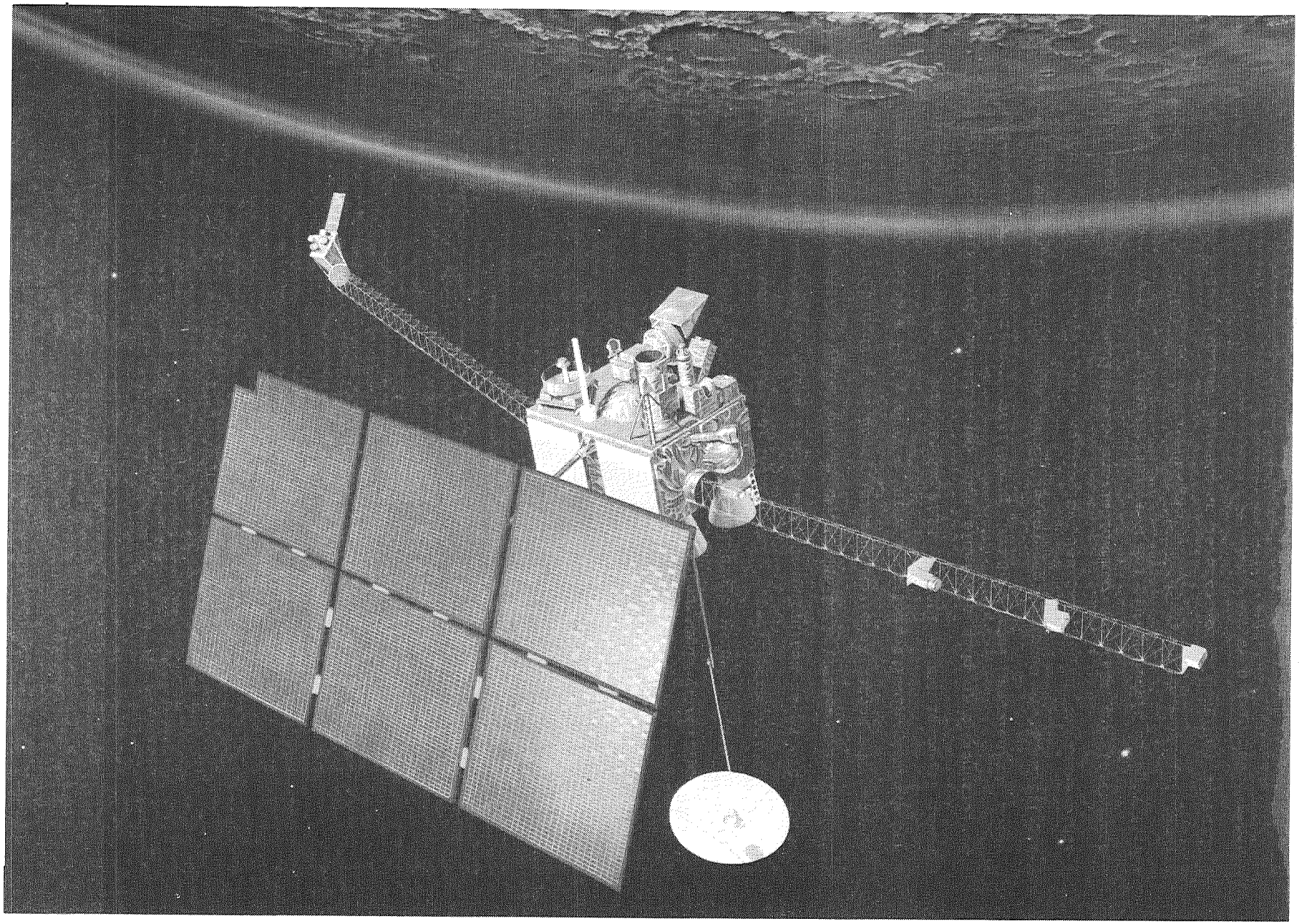


GALILEO PLANETARY SPACECRAFT

MARS OBSERVER PLANETARY SPACECRAFT

The Mars Observer spacecraft will be used for the first planetary observer mission. A year after its September 1992 launch, it will enter a low-altitude mapping orbit to make continuous observations of the planet's surface and atmosphere over a full Martian year. The spacecraft instruments will include reflectance, emission and gamma ray spectrometers, a radiometer, an altimeter, a camera, a magnetometer and radio science instrumentation.

The Mars Observer spacecraft will utilize both bipropellant (NTO/MMH) and monopropellant (hydrazine) engines. The bipropellant engines, built by Atlantic Research, are planned for the trajectory control maneuvers, orbit insertion and pitch and yaw control. There are four 490-N engines and four 22-N engines in the bipropellant system. The monopropellant engines, built by Rocket Research, are planned for reaction wheel desaturation, orbit trim maneuvers and roll control. There are eight 4.45-N engines and four 0.9-N engines in the monopropellant system. The Mars Observer spacecraft weighs approximately 2500 kg at the beginning of life, with about 57 percent of that weight being propellants.



MARS OBSERVER PLANETARY SPACECRAFT

CRAF SPACECRAFT

Following its launch in 1995, the Comet Rendezvous Asteroid Flyby (CRAF) spacecraft will fly past one asteroid on its journey to rendezvous with a comet. As the spacecraft accompanies the comet on its orbit around the sun, cameras and instruments will record the onset of comet activity as the comets dirty ice nucleus warms and creates a glowing atmosphere of gas and ions around it.

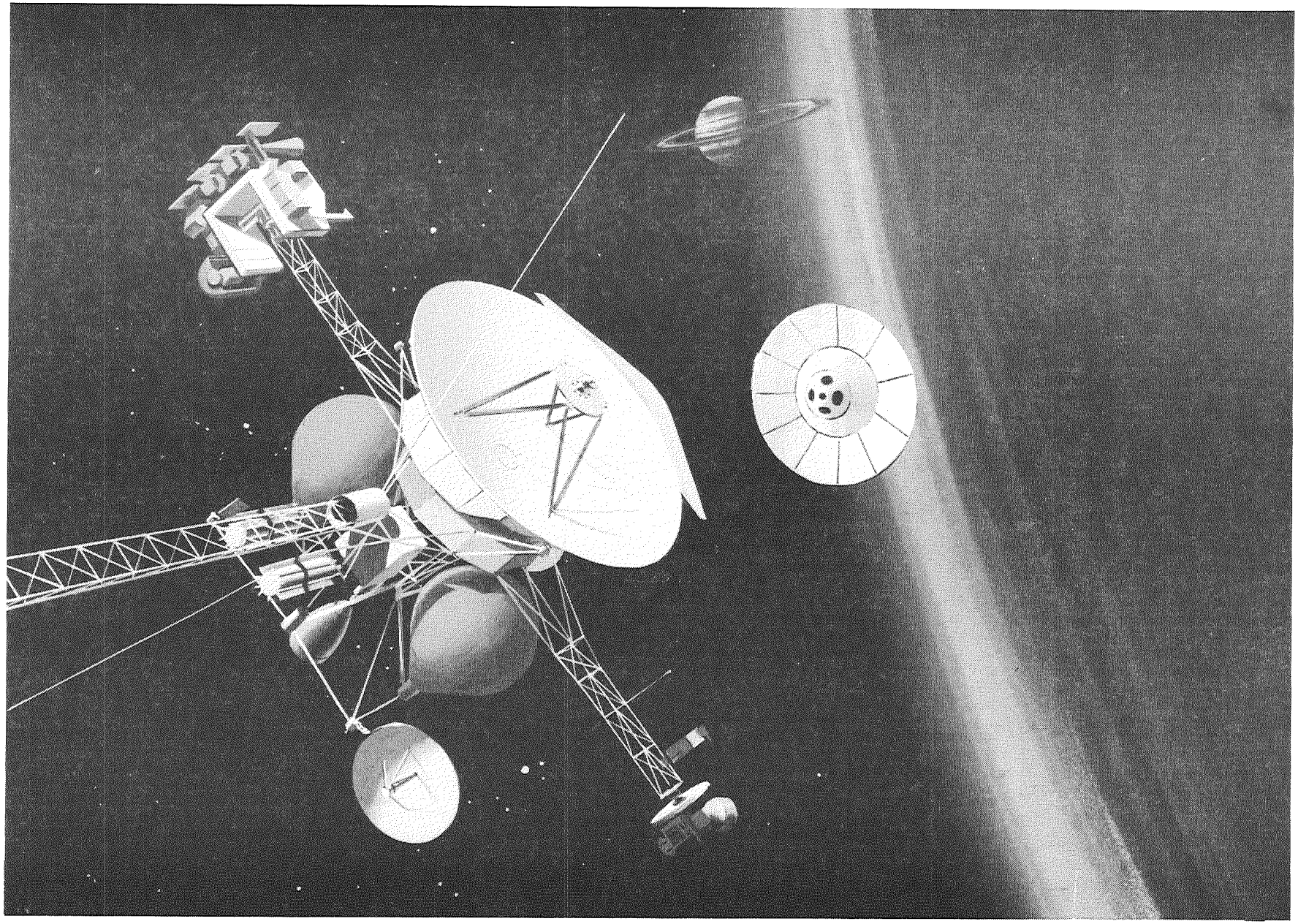
The CRAF spacecraft will utilize both bipropellant (NTO/MMH) and monopropellant (hydrazine) engines. The single 490-N bipropellant engine is built by Marquardt, while the sixteen 0.5-N monopropellant engines are built by MBB-ERNO. The CRAF spacecraft weighs approximately 5400 kg at the beginning of life, with about 70 percent of that weight being propellants.

CRAF SPACECRAFT

CASSINI SPACECRAFT

The Cassini spacecraft is designed as the second Mariner Mark II spacecraft and is scheduled for launch in April 1996. As a joint NASA-ESA orbiter and probe mission to the Saturian system, the probe will be delivered to Titan, where it will make measurements of the atmosphere as it descends to the surface. Following delivery of the probe, the orbiter will conduct as intensive four-year investigation of Saturn's atmosphere, ring system, the icy satellites, the magnetosphere, Titan's upper atmosphere and, using an onboard radar, the surface of Titan.

The Cassini spacecraft will be almost an exact duplicate of the CRAF spacecraft, except the propellant load will be reduced by about 300 kg.



CASSINI SPACECRAFT

HYDRAZINE RESISTOJETS (TRW)

The electrically augmented hydrazine thruster is an example of a wall-heated thruster. Hydrazine resistojets built by TRW are operational on Ford Aerospace INTELSAT V communications satellites. The INTELSAT V satellite weighs approximately 1170 kg at beginning of life with a solar array power of 1800 W. The INTELSAT uses four hydrazine resistojets for N-S stationkeeping. These resistojets produce a thrust of 0.49 - 0.22 N, an exhaust velocity of 2.9 km/s and require a power input of 550 W - 250 W. Ongoing research at TRW is directed at increased thruster life and performance. Exhaust velocities as high as 3.3×10^3 m/s and life in excess of 2.6×10^3 Ns have been demonstrated.

JPL

HYDRAZINE RESISTOJETS BUILT BY TRW ARE OPERATIONAL ON FORD AEROSPACE INTELSAT V COMMUNICATION SATELLITES

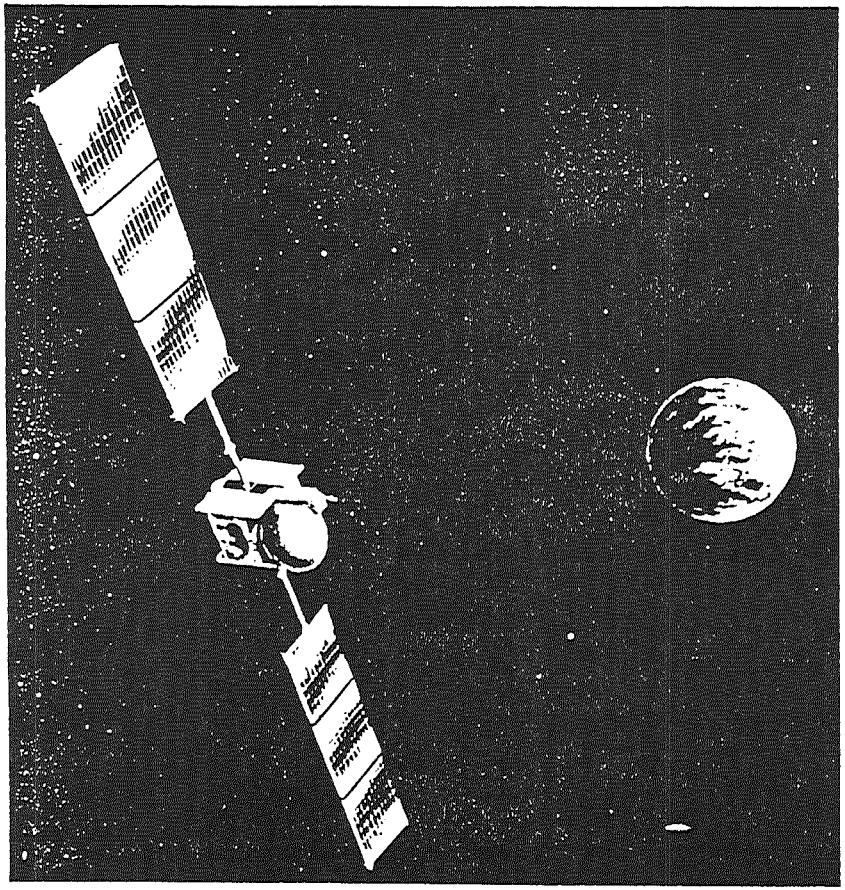
INTELSAT V
- 3 AXIS STABILIZED
- 10 YEAR LIFE

• THRUST	0.49-0.22 N
• EXHAUST VELOCITY	2.9×10^3 ms^{-1}
• POWER	550-250 W

HYDRAZINE RESISTOJETS (ROCKET RESEARCH)

Electrically-augmented hydrazine thrusters manufactured by Rocket Research are currently operational on RCA SATCOM, G-Star and Spacenet communication satellites. The G-star satellite shown in the photo weighs 670 kg at beginning of life with a solar array power of 1065 W. The SATCOM-K satellite weighs 985 kg at beginning of life with 2000 W of power. The Rocket Research thruster produces a thrust of 0.36 - 0.18 N and exhaust velocities of 2.74×10^3 - 2.98×10^3 m/s for an electrical power input of 500 - 300 W. This represents a 30 percent increase in performance over that available from conventional hydrazine thrusters which translates into reduced propellant requirement or increased satellite life. Ongoing research at Rocket Research is directed at the achievement of higher exhaust velocities and higher thrust. Future commercial and military communications satellites will be larger and will require higher thrust for N-S stationkeeping. In work sponsored by INTELSAT, Rocket Research has designed and is fabricating a 2.0-N thruster designed to operate at an exhaust velocity of 3.04×10^3 m/s and a power of 2.0 kW.

JPL HYDRAZINE RESISTOJETS BUILT BY ROCKET RESEARCH ARE OPERATIONAL ON RCA SATCOM, G-STAR AND SPACENET COMMUNICATION SATELLITES

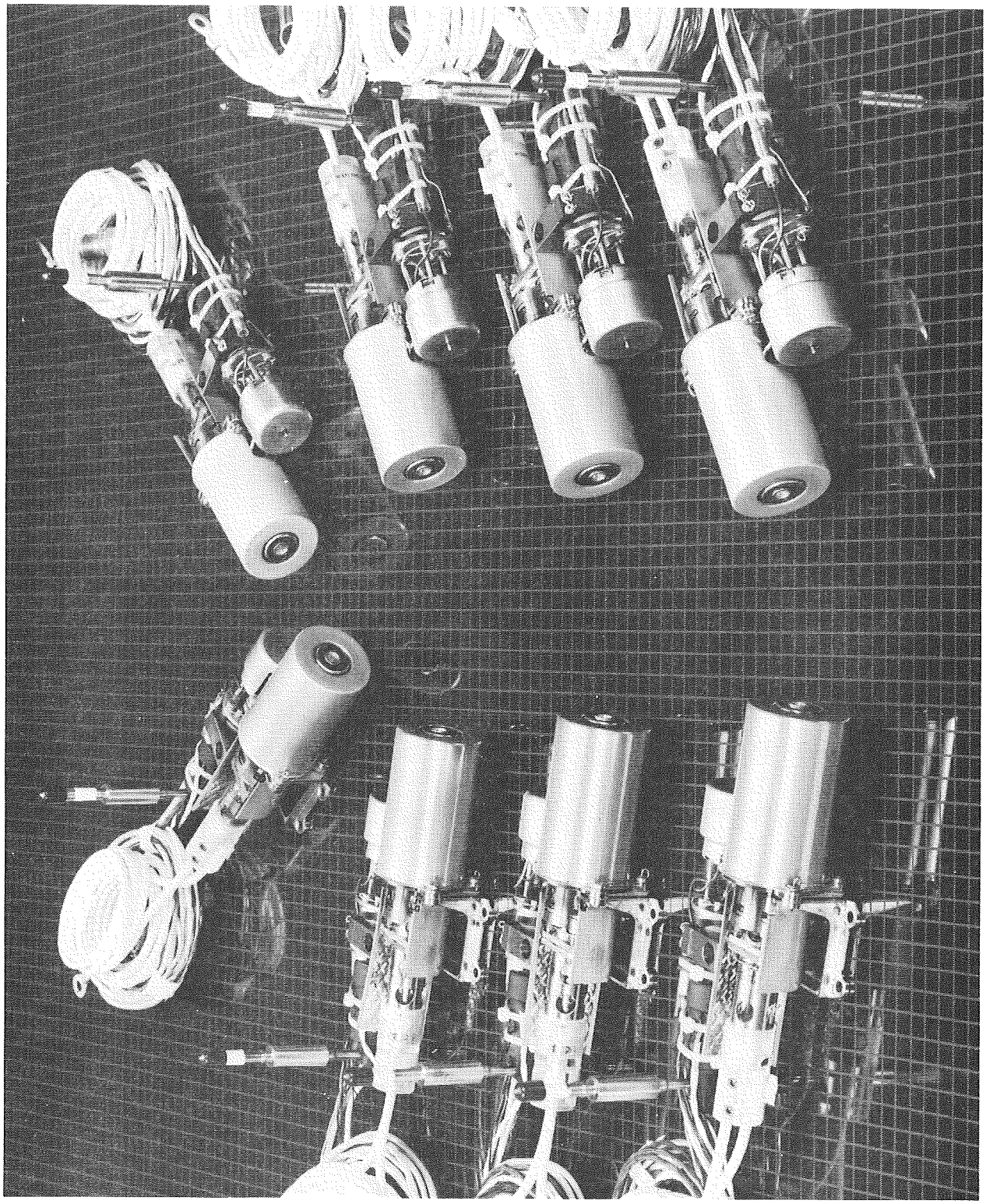


G-STAR

- 3 AXIS STABILIZED
- 10 year LIFE

THRUST	0.36-0.18 N
EXHAUST VELOCITY	$2.7\text{-}3.0 \times 10^3$ ms^{-1}
ELECTRICAL POWER	500-300 W

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HYDRAZINE RESISTOJETS

LOW-POWER ARCJET SYSTEM

Recently, low-power arcjets have been studied for application on the N-S stationkeeping function on communication satellites. Research programs at NASA LeRC, Rocket Research and Purdue University have led to a better understanding of energy loss mechanisms and arc stability. Rocket Research has demonstrated performance of 510 - 550 lbf-s/lbm for a low-power hydrazine arcjet operating at chamber pressures of 60 - 70 psia, thrust levels of 0.045 - 0.052 lbf and an input power of 1800 W. The Rocket Research low-power arcjet system (MR-508) is scheduled for use on a GE AstroSpace spacecraft for the Telstar IV communication satellite. The satellite is scheduled for a 1992 launch.

LOW-POWER ARCJET SYSTEM

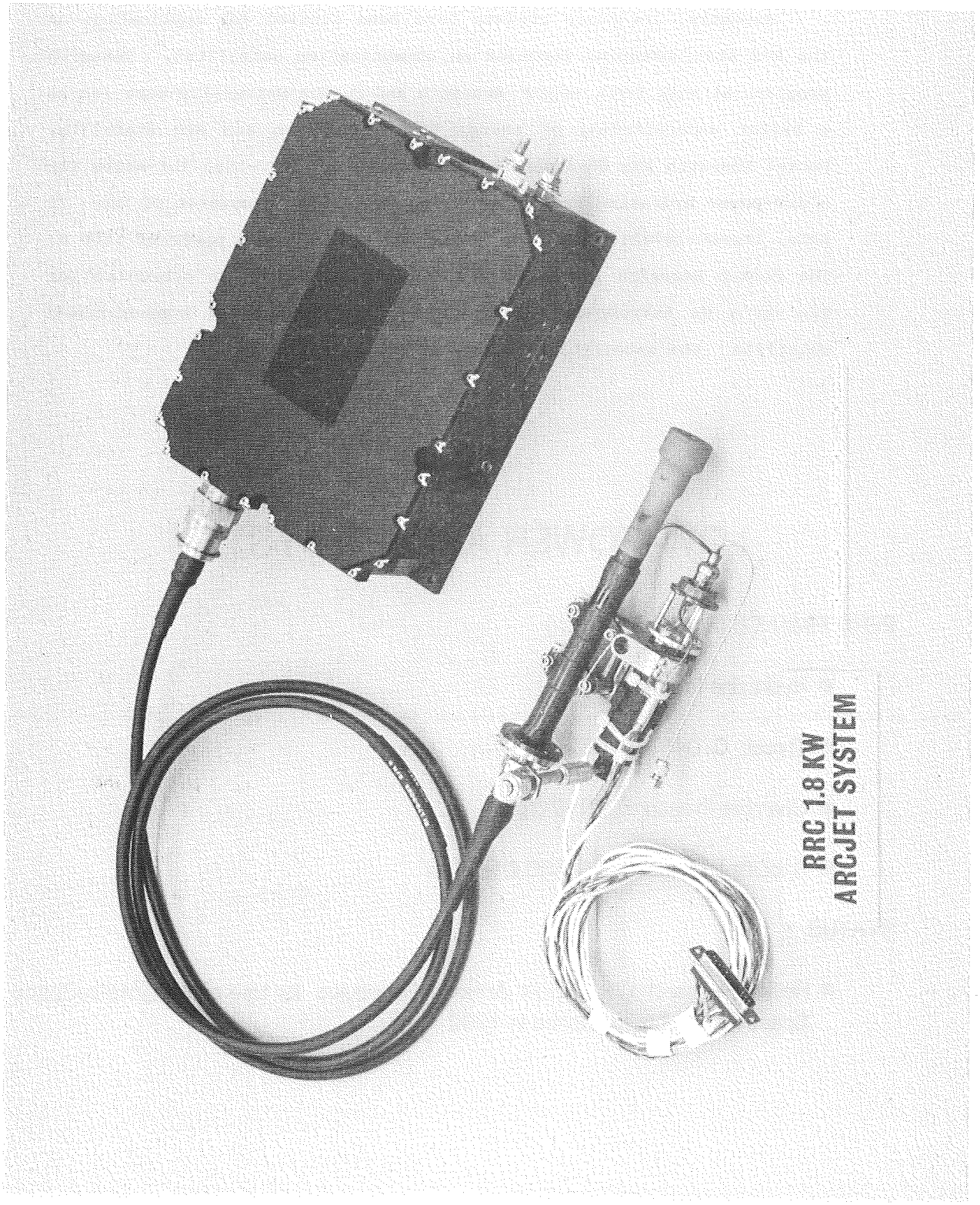
PERFORMANCE (Rocket Research)

- Hydrazine Propellant
- Thrust: 0.045 - 0.052 lbf
- Chamber Pressure: 60 -70 psia
- Specific Impulse: 510 - 550 lbf-s/lbm

STATUS

- Rocket Research Low-Power Arcjet is Scheduled for Use on a GE AstroSpace Spacecraft to be Launched in 1992

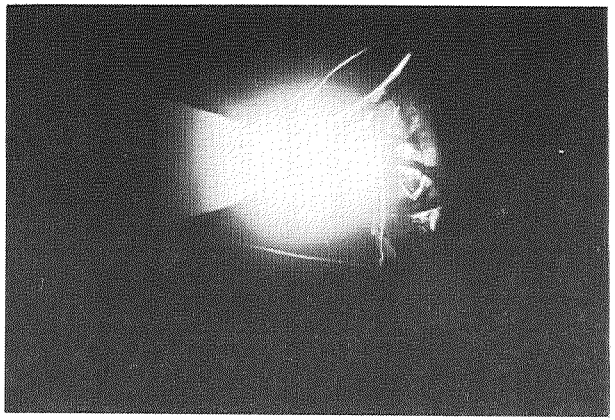
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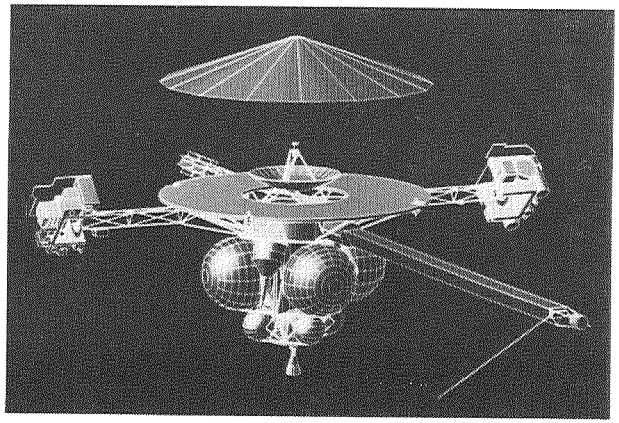


HIGH-PERFORMANCE ENGINE

Conventional radiation-cooled bipropellant engines utilize disilicide-coated columbium thrust chambers which have a nominal operating temperature of 2400 F. Work on high temperature rhenium thrust chambers was begun at JPL in the mid 1970's as part of the High Energy Propulsion System (HEPS) program. Rhenium permits thrust chamber operating temperatures of greater than 4000 F; however, rhenium has a low oxidation resistance. Since the mid 1980's, NASA LeRC has had a research program looking at high temperature materials and coatings for thrust chambers. Recently, a feasibility demonstration effort has been conducted at Aerojet TechSystems under JPL contract to see if this high performance engine technology could be made available for CRAF/Cassini missions. The demonstration was conducted using a 445-N bipropellant engine (NTO/MMH) with a thrust chamber fabricated from iridium-coated rhenium. The increased bipropellant engine performance (326 lbf-s/lbm) offered by this technology reduces the injected mass requirement for the CRAF mission by more than 600 kg compared with conventional bipropellant engine performance (308 lbf-s/lbm). Although the high performance bipropellant engine technology is not currently being pursued for the CRAF mission, due to lower propulsion requirements, this technology is still being pursued by NASA LeRC and the U. S. propulsion industry.

- JPL CONTRACT BUILDS ON LeRC TECHNOLOGY PROGRAM
- I_{SP} IMPROVED TO 326 $lb_f \cdot s/lb_m$
- REDUCES PROPULSION SYSTEM WET MASS ALMOST 600 kg
- ENABLES CRAF TO ADD 3 EXPERIMENTS





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OSSA

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DUAL MODE PROPULSION SYSTEM

A dual mode propulsion system offers advantages over conventional spacecraft propulsion systems. The dual mode propulsion system consists of a bipropellant engine (NTO/hydrazine) for the main propulsion function and a monopropellant (hydrazine) engine for the attitude control functions. This arrangement permits the bipropellant and monopropellant engines to share common propellant tankage. The bipropellant engine (NTO/hydrazine) offers higher performance than conventional bipropellant engines (NTO/MMH). As a result, the dual mode propulsion system offers a substantial mass savings over conventional systems. The new propellant combination (NTO/hydrazine) offers plume contamination advantages over conventional bipropellant systems (NTO/MMH) due to the absence of carbon in the fuel. The use of monopropellant hydrazine for attitude control functions also leads to lower contamination.

TRW has been developing a dual mode propulsion system for spacecraft application. The bipropellant engine (NTO/hydrazine) has demonstrated a performance of 313 lbf-s/lbm compared with conventional bipropellant engines with a performance of 308 lbf-s/lbm. The TRW dual mode propulsion system is scheduled for use on the GE AstroSpace Series 5000 spacecraft for Canadian, SES and Intelsat-K communication satellites. The first launch is expected to be the SES satellite in January 1991.

DUAL MODE PROPULSION SYSTEM

DESCRIPTION

- Main Propulsion Uses Bipropellant Engine (NTO/hydrazine)
- Attitude Control Function Uses Monopropellant Engine (hydrazine)

ADVANTAGES

- Higher Bipropellant Engine Performance, Lower Contamination Potential
- Common Propellant Tankage, Lower Propulsion System Mass

STATUS

- TRW Developing Dual Mode Propulsion System
- Bipropellant Engine Demonstrated 313 lbf-s/lbm
- TRW Dual Mode Propulsion System Scheduled for Use on a GE AstroSpace Spacecraft to be Launched in 1991

SUMMARY

- Propulsion system performance has high leverage for many future missions because of large propellant mass requirements. Relatively small performance improvements can translate into large increases in payload and science return.

- Contamination control becomes more important as science instruments become more sensitive. This places more emphasis on exhaust plume contamination control.

- The need for reliable operation and long life places increased importance on health monitoring and control of spacecraft propulsion systems.

- The need for accurate spacecraft pointing and control increases the need for small impulse-bit thrusters.

PROPULSION SYSTEMS OPTIONS-

NEXT GENERATION SYSTEMS

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PRESENTATION 1.3.1

SHUTTLE DERIVATIVES - MANNED

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SHUTTLE DERIVED MANNED TRANSPORTATION SYSTEMS

SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM
PENNSYLVANIA STATE UNIVERSITY
JUNE 1990

WAYNE L. ORDWAY
SYSTEMS ENGINEERING DIVISION
NASA JOHNSON SPACE CENTER
HOUSTON, TX

SHUTTLE DERIVED MANNED TRANSPORTATION SYSTEMS

Wayne L. Ordway
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Houston, TX

Abstract

Shuttle derivatives have been under study by the National Aeronautics and Space Administration (NASA) for a number of years. With Space Station Freedom and the Lunar/Mars Initiative established as national objectives, the demand for access to Earth orbit is accelerating. These objectives have resulted in efforts to address additional launch requirements that must be met as we approach the turn of the century. Among the top level requirements are increased safety, higher reliability, lower cost, and the need for heavy lift launch capability. To satisfy these requirements, some of the largest technology demands will be placed upon the propulsion systems. This paper will present Shuttle derived manned concepts and will discuss the associated propulsion issues which arise from the top level requirements . These concepts are presented in terms of an overall architecture which can be achieved with modest up-front development.

Introduction

Space Shuttle derivative studies conducted over the past decade have primarily emphasized cargo vehicles. Shuttle Evolution assessments initiated in 1988 are attempting to address the corresponding issues for manned transportation systems. This paper will discuss some Shuttle derivatives with particular application to manned missions, though cargo delivery will be addressed in order to describe an architectural solution. Consideration of all three fundamental Shuttle hardware elements, the External Tank (ET), boosters, and Orbiter is essential to the evolution of an architecture which will meet long term requirements.

The primary goals for the next manned transportation system are to achieve increased reliability and safety, lower operational costs, and increased operational capability. As historically demonstrated throughout the aircraft and aerospace industry, such needs can be satisfied efficiently by introducing block upgrades to the elements of the system which have operational shortcomings. Shuttle operational experience has identified one of the prominent elements influencing reliability, safety, and cost to be the vehicle propulsion systems. The challenge of meeting the goals for the next generation systems will impose direct requirements upon the technologies and philosophy to be applied to development of new and/or modified propulsion systems. These requirements, to a large extent, will be imposed on both the manned and unmanned transportation system elements.

Launch Requirements

The civilian space requirements are formulated in the Civil Needs Data Base (reference 1) and are augmented by the requirements postulated in the Human Exploration Study performed by NASA in the Fall of 1989 (reference 2). Although preliminary, these sources enable determination of the fundamental launch requirements. The deliverables can be broadly categorized into the transportation of personnel, hardware, and propellant.

Extending human presence in space will require a considerable increase in the crew rotation capability beyond the present maximum of 70 crew members per year. This rate is based upon a Shuttle capability of 14 flights per year and 2 crew/5 passengers per flight. Projected requirements approach a rotation rate of 90 passengers per year in the 2010 time period with a Lunar/Mars initiative (figure 1). Increasing the crew capacity of the Shuttle to 10 (2 crew/8 passengers) is considered a viable option and becomes a basic requirement for the Shuttle derived system described in this report.

Requirements for cargo delivery must be examined for both hardware and propellant delivery since the two payload types can result in different delivery systems. For a typical Lunar mission, based on the requirements in reference 2, the total system mass in low Earth orbit (LEO) is on the order of 450K lbs for an aerobraked, fully fueled LOX/LH2 transfer system. The capability for a direct launch, Lunar mission is highly desirable for an early Lunar program and would also enable reasonable means of initiating more aggressive missions (e.g. Mars). This goal establishes an upper, lift capability requirement of 450K lbs on the derived launch system. The Lunar mission LEO mass of 450K lbs breaks out into 300K lbs of required propellant and 150K lbs of hardware. These masses are representative of re-supply requirements for hardware and propellant for projected Lunar missions. Once the reusable, space based hardware is in place, however, propellant will become the dominant commodity. Consideration of these projected lift requirements has led to study of modular, heavy-lift transportation systems with payload capabilities up to 450K lbs.

Candidate Evolution Strategy

To address the goals of lower operational costs and increased capability for the next manned transportation system, an evolutionary strategy has been proposed which utilizes Shuttle derived hardware elements and draws upon the lessons of Shuttle operational experience (reference 3). The basic elements comprising the evolutionary architecture are: 1) an External Tank (ET) derived core stage, 2) a liquid rocket booster (LRB) system, and 3) a Block-II Orbiter lacking the main propulsion system.

A core stage consisting of a modified ET with an integrated main propulsion system has been previously studied (references 4,5). Figure 2 illustrates a candidate concept which is configured with three Space Shuttle Main Engines (SSME) and an optional propulsion return module. Standard SSMEs, to be operated at 100 percent thrust levels,

were baselined in this design in consideration of the planned improvements and the extensive operating experience and reliability which will have been achieved by the time the evolved systems become operational. To provide capability for the orbital insertion and maneuvering requirements typical of propellant delivery missions, provision is also made for a separate orbital maneuvering/reaction control system. The derivative concepts under consideration are intended to remain flexible to the incorporation of new, low cost propulsion systems which become available.

Based upon studies performed in 1988-89 (references 6,7), a new LOX/LH2 liquid rocket booster (LRB) system is a favored candidate for the evolution architecture. With the LRB concept shown in figure 3, the system's payload capability to LEO can be extended to 65-70K lbs. Among the many desirable attributes of this system are common propellant and engine systems, potential redundancy for engine out, abort options, environmentally clean exhaust, improved ground processing and safety, and growth potential. Additionally, the LRB has considerable synergism with heavy-lift launch vehicle concepts and with alternate access options such as the Personnel Launch System (PLS). The low cost, reliable propulsion systems developed for the LRBs may also have application to long-term evolution concepts of a "Shuttle-II" system incorporating fly-back boosters.

To address the requirement for increased crew capacity, a "Block-II" Orbiter is proposed with an enlarged crew compartment designed to accommodate a crew of ten. Removal of the main propulsion system from the Orbiter, enabled with a core stage concept, is the next major modification which offers several advantages. First, it separates the launch function from the spacecraft, with an associated reduction in vehicle complexity. Second, it provides the potential for increased operational capability. The available volume from removal of the propulsion system could house additional orbital maneuvering system propellant and the Orbiter weight reduction could translate into down payload capability. Additional enhancements which have been defined in recent Shuttle Evolution studies are included in the "Block-II" concept. These enhancements address a variety of vehicle subsystems and are designed to achieve the top level transportation system goals. The "Block-II" Orbiter concept is illustrated in figure 4.

The complete, Shuttle derived launch vehicle concept is depicted in figure 5 along with the estimated performance capability which results from enhancement weight changes. Performance capability for the derived Orbiter concept, however, is not considered the primary goal. If it is assumed that cargo delivery will be performed to a large extent by unmanned launch systems, performance capability can be traded for increased margins enabling the "Block-II" Orbiter to emphasize enhanced crew capability and on-orbit operations.

The described modifications to the Shuttle elements produce a manned transportation system which offers flexible architecture options. Elements from this system can be used to provide alternate access with a Personnel Launch System as well as substantial heavy-lift payload delivery with cargo and propellant launch vehicles. Modular, heavy-lift launch vehicle concepts incorporating a stretched core stage and 6-8 LRBs can be configured to meet a single launch Lunar mission cargo

requirement of 450K lbs. This vehicle can satisfy Lunar mission needs with minimum required on-orbit assembly and check-out and also provides reasonable capability for initiation of a Mars program. The overall evolution strategy requires no technology breakthroughs and is capable of meeting a wide range of requirements well into the next century. An illustration of the fundamental architecture is presented in figure 6.

System Requirements

Achieving the top level goals of increased reliability and safety, and lower operational costs for the next space transportation systems will require that an integrated systems engineering approach be employed throughout the design. The fundamental requirements placed upon the vehicle subsystems must be derived to optimize the overall system goals. With the substantial cost which will be associated with future systems and payloads, the reliability expectations for unmanned cargo vehicles have become as demanding as for the manned vehicles. In order to assess how the requirements for these vehicles differ, the subject of man rating must be addressed.

A man-rated system is defined to be one for which all elements are designed with the highest possible reliability, including the required escape system or safe haven. The philosophy applied to these systems emphasizes simple designs whenever possible and the use of only proven technology. Where application of new technologies appears beneficial, technology development programs should precede in order to evaluate reliability. A basic set of guidelines has been established which constitute design criteria for the man-rating of space systems (reference 8). The design emphasis prescribed for the system generally dictates the extent to which these guidelines are applied (figure 7). A summary of the man rating design guidelines is presented in figure 8.

One of the foremost criteria unique to man-rated systems is the requirement for a crew escape system. Design studies being conducted within NASA are evaluating several approaches for ensuring crew safety in the next manned space vehicles. Crew escape options under consideration range from basic ejection concepts to intricate crew escape modules designed to survive the most catastrophic failure. Implicit in the requirement for crew escape provisions is a corresponding requirement for fault detection capability. Accurate and reliable means for sensing and isolating critical hazards is fundamental to crew safety and abort flexibility and is an essential requirement applicable to all critical systems for man-rated vehicles.

With regard to vehicle propulsion systems, an issue which arises specifically from man-rating considerations is the requirements on engine throttling capability imposed for ascent g-limiting and abort criteria. Engine throttling requirements need to be evaluated and set from a vehicle-level assessment of capability versus system complexity. Imposing throttling constraints based upon propulsion system considerations alone may not properly address the top level goals for the vehicle. Another issue with implications to engine throttling is the desire for engine-out capability. This approach to improving overall reliability will introduce a minimum throttle-up requirement upon the propulsion system. Fundamental to the engine-out design

philosophy is an assumed low probability of catastrophic engine failure. This places a basic requirement on the engine design to emphasize benign failure modes, in which other elements are not damaged by a failure, to the greatest extent possible. Approaches to engine design which minimize the potential for catastrophic failures have been identified from evaluation of historical engine failures (reference 9). In consideration of these many critical functions to be performed through propulsion system throttling, minimizing the failure potential of the throttling function in itself will be of utmost importance.

The remaining propulsion issues address the top-level goals of high reliability and low cost and are considered to be equally as important for unmanned systems as for manned systems. Ensuring high reliability for the next transportation systems may favor new approaches to propulsion system design. An example of one such approach is integrated system designs with sharing of components (reference 10). New and innovative design approaches need to be studied to substantiate their benefit potential. Regardless of the design approach, however, there are common propulsion requirements which can be discussed. The system and its components will be required to be fault tolerant. Another basic requirement will be the need for a comprehensive test program designed to verify functional reliability and establish system failure limits. The system's limitations and safety margins should be determined through off-limits testing including tests-to-failure to demonstrate the failure modes and effects. The capability for on-board, automated check-out and verification is also a desirable provision of future propulsion systems. In general, a requirement for some degree of propulsion system health monitoring and control will need to be specified.

In consideration of the lessons learned through Shuttle operational experience, a clear requirement for future propulsion systems will be improved maintainability and minimized hazardous operations. As shown in figure 9, the Shuttle's main propulsion system is responsible for a significant percentage of the Shuttle's operational processing time. Emphasis placed upon simplicity and accessibility during the design process can translate directly to reduced propulsion system operational costs. A summary of the issues and requirements identified for next generation propulsion systems is presented in figure 10.

Conclusion

An architectural strategy which utilizes Shuttle derived elements and a new LRB system appears a viable approach to achieving the goals of higher reliability, lower operational costs, and increased capability for the next manned transportation system. Evolution with a "Block-II" system offers the potential benefits of reduced risk and lower up-front development costs. The foreseen requirements for vehicle propulsion systems predominantly address the need for fault tolerance and health monitoring capability. High reliability is an expectation for both manned and unmanned systems. Specific requirements for propulsion throttling capability may arise for manned vehicles and will need to be derived on the basis of the vehicle requirements.

References

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2. Human Exploration Study Requirements, M. Craig/NASA-JSC, September 8, 1989.
3. Evolutionary Transportation Concepts, C. Teixeira/NASA-JSC, April, 1990.
4. Shuttle Derived Vehicles Technology Requirements Study, NAS8-34183, Martin Marietta Co., 1982.
5. Shuttle Derived Cargo Launch Vehicle Concept Evaluation Study, NAS8-34599, Boeing Co., 1982.
6. Liquid Rocket Booster Study, General Dynamics Co., NAS8-31737, May 18, 1988.
7. Liquid Rocket Booster for the Space Transportation System Study, Martin Marietta Co., NAS8-37136, March 1989.
8. Guidelines For Man Rating Space Systems, M.P. Cerimele, NASA-JSC-23211, September, 1988.
9. Propulsion System Integration for The Advanced Launch System, AIAA-89-2747, July, 1989.
10. Operationally Efficient Propulsion Systems Study, NAS10-11568, Rockwell International, April 1990.

EVOLVING REQUIREMENTS

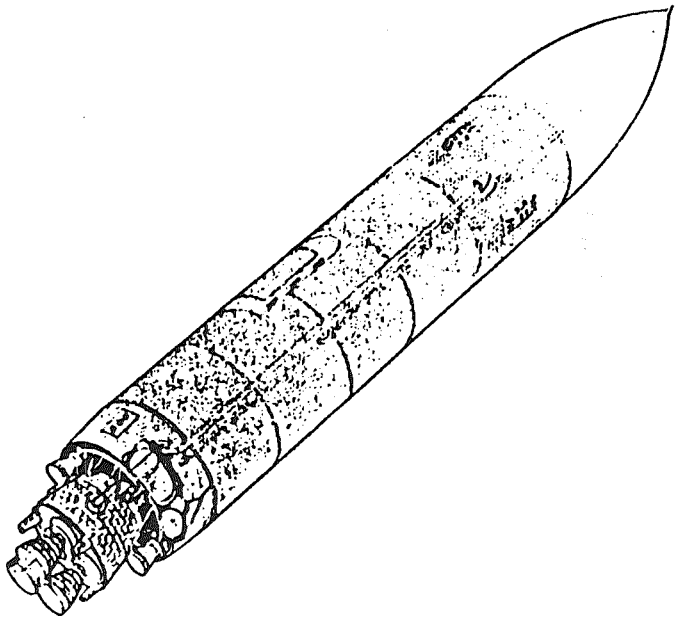
- EXPANDED HUMAN PRESENCE IN SPACE
- ORDER MAGNITUDE INCREASE IN LIFT CAPABILITY
- PROPELLANT (BULK) DELIVERY
- DOWN PAYLOAD

Figure 1

CORE STAGE

(EXTERNAL TANK DERIVATIVE)

- BENEFITS
 - IN - LINE PROPULSION
 - ENABLES SIMPLER ORBITER
 - SEPARATES LAUNCH PROPULSION FROM SPACECRAFT
 - ENABLES MODULAR LAUNCH SYSTEM



CORE STAGE WEIGHT SUMMARY

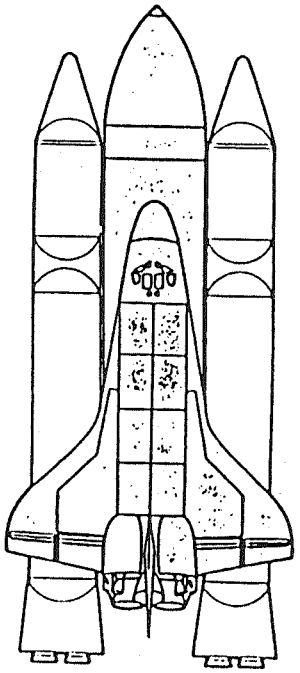
	EXPENDABLE 3 SSME P/A	WITH P/A MODULE	WITH ORBITAL INSERTION CAPABILITY
BASELINE ET (ET-41)	66588	66588	66588
AL-LI ET	-8000	-8000	-8000
ENGINES, STRUCTURE FEED LINES, ETC	32000	32000	32000
DELTA WEIGHT FOR THRUST LOADS	7710	7710	7710
AFT SKIRT	6360	6360	6360
SUBSYSTEMS, INCL AVIONICS, TVC	8929	8929	8929
OMS/RCS	--	--	8684
P/A MODULE	--	9929	9929
SUBTOTAL	113587	123516	132200
10% MARGIN **	116687	127609	136293

**APPLIED TO NEW HARDWARE ONLY

Figure 2

LIQUID ROCKET BOOSTER

- BENEFITS
 - ENHANCED SAFETY
 - SHUT DOWN CAPABILITY
 - ADDITIONAL ABORT MODES
 - PRE-LIFT-OFF PERFORMANCE VERIFICATION
 - ENVIRONMENTALLY CLEAN
 - EFFICIENT LAUNCH OPERATIONS



LRB

Length (ft)	178
Diameter (ft)	18
Booster dry weight (lb)	122,000
Booster gross weight (lb)	821,000

Features

- LH2/LO2 propellants
- 2219 aluminum tankage
- New low-cost, pump-fed engines
- 4 engines per booster
- Expendable (engines may be recovered)
- Existing technologies

Figure 3

ENGINELESS ORBITER

ORBITER WEIGHT SUMMARY

	WEIGHT, LBS
OV 103, STS-37*	228000.00
TOP 10 ENHANCEMENTS	
ELECTROMECHANICAL ACT.	-5000.00
ADV. FUEL CELLS	--
INTEGRATED OMS/RCS,	-1590.00
OMS POD REDESIGN	
IMPROVED VERNIERS	--
IMPROVED AVIONICS	-1500.00
INCREASED CREW	1350.00
AUTOMATED ORBITER	--
COMPOSITE FLAP, ELEVONS, OMS POD	-1508.00
ADVANCED TPS	-1580.00
EXTENDED NOSE GEAR	--
INTEGRATED THERMAL CONTROL SYSTEM	TBD
BLOCK-II SSME	750.00
	218922.00
LESS PROPULSION SYSTEM	-40800.00
TOTAL	178122.00
PLUS CREW ESCAPE MODULE	15000.00
*TRANSATLANTIC ABORT CONDITION	193122.00

- BENEFITS
 - INCORPORATES "TOP 10" ENHANCEMENTS
 - MUCH SIMPLER, LIGHTER
 - QUICKER TURNAROUND AT KSC

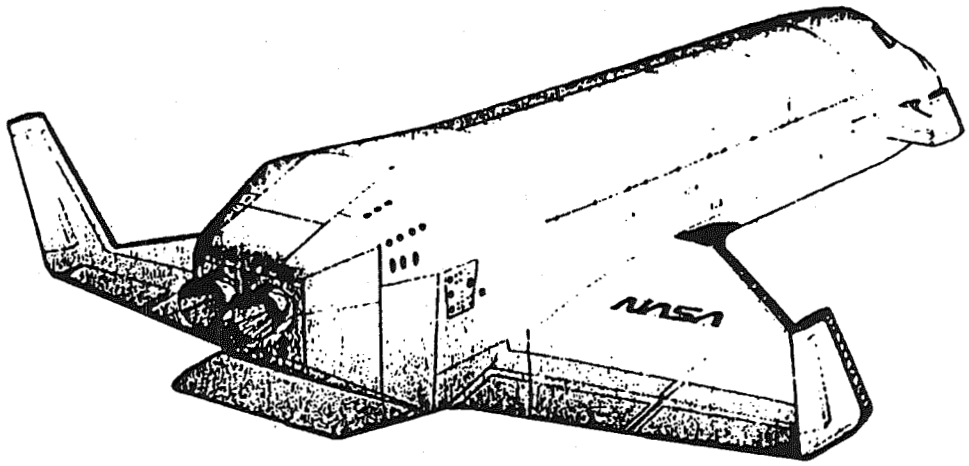


Figure 4

INTEGRATED LAUNCH STACK

PERFORMANCE IMPACTS
DUE TO WEIGHT CHANGES

	PAYLOAD, LBS
NSTS BASELINE	65000
BLOCK II ORBITER (WITH CEM)	37840
ET CORE STAGE	-70705
LRB	20000
NET PAYLOAD	52135 (67135 WITHOUT CEM)
MARGIN ENHANCEMENT (CANDIDATES LIST)	
LOWER DYNAMIC PRESSURE STANDARD I-LOAD	-5000
100% SSME	-4000
PERFORMANCE TRADED FOR MARGINS, LOWER COST	-9000
NET PAYLOAD	43135 (58135 WITHOUT CEM)

Figure 5

Aggressive SSV Evolution Option

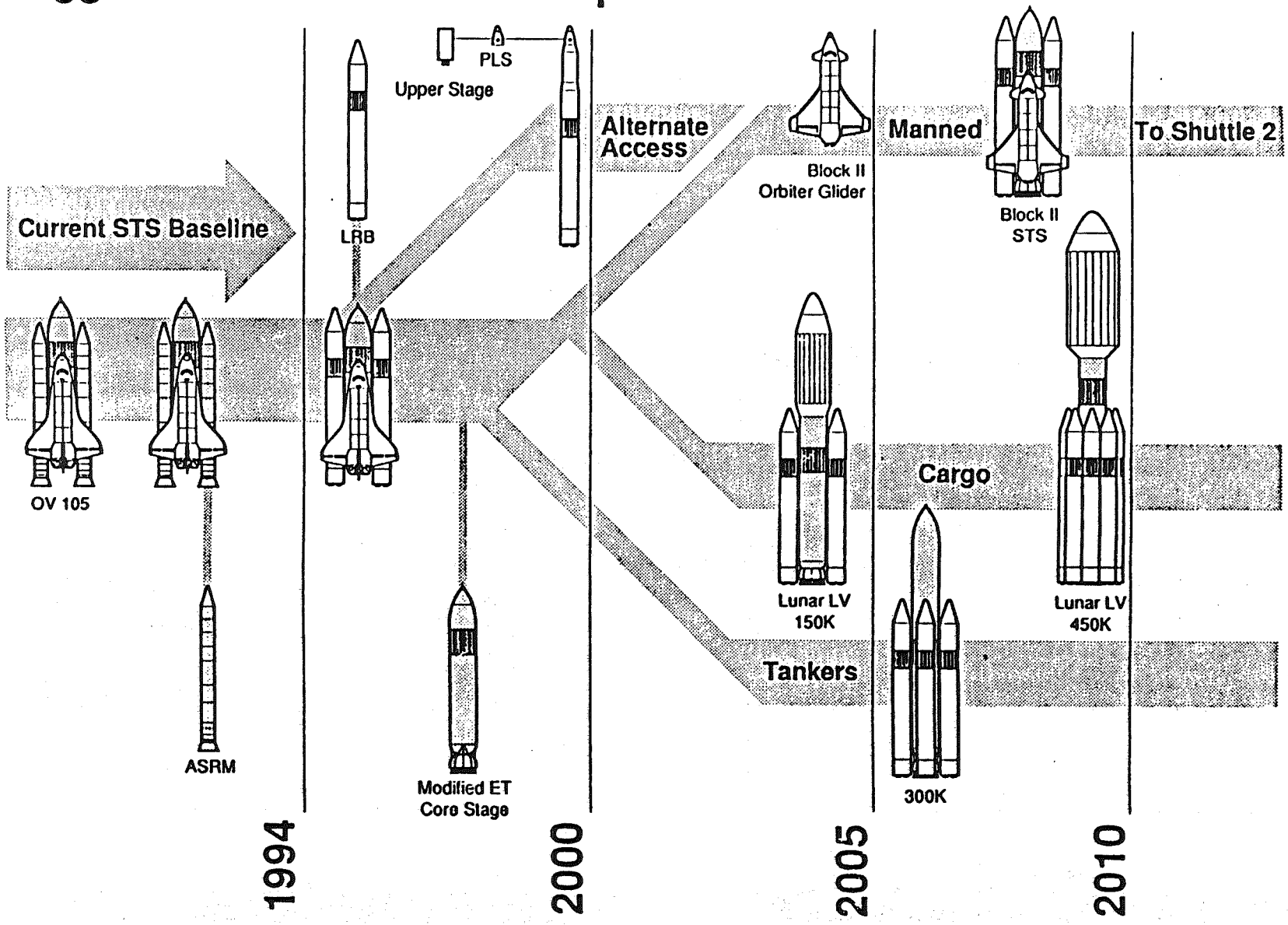


Figure 6

MISSION RISK LOGIC DIAGRAM

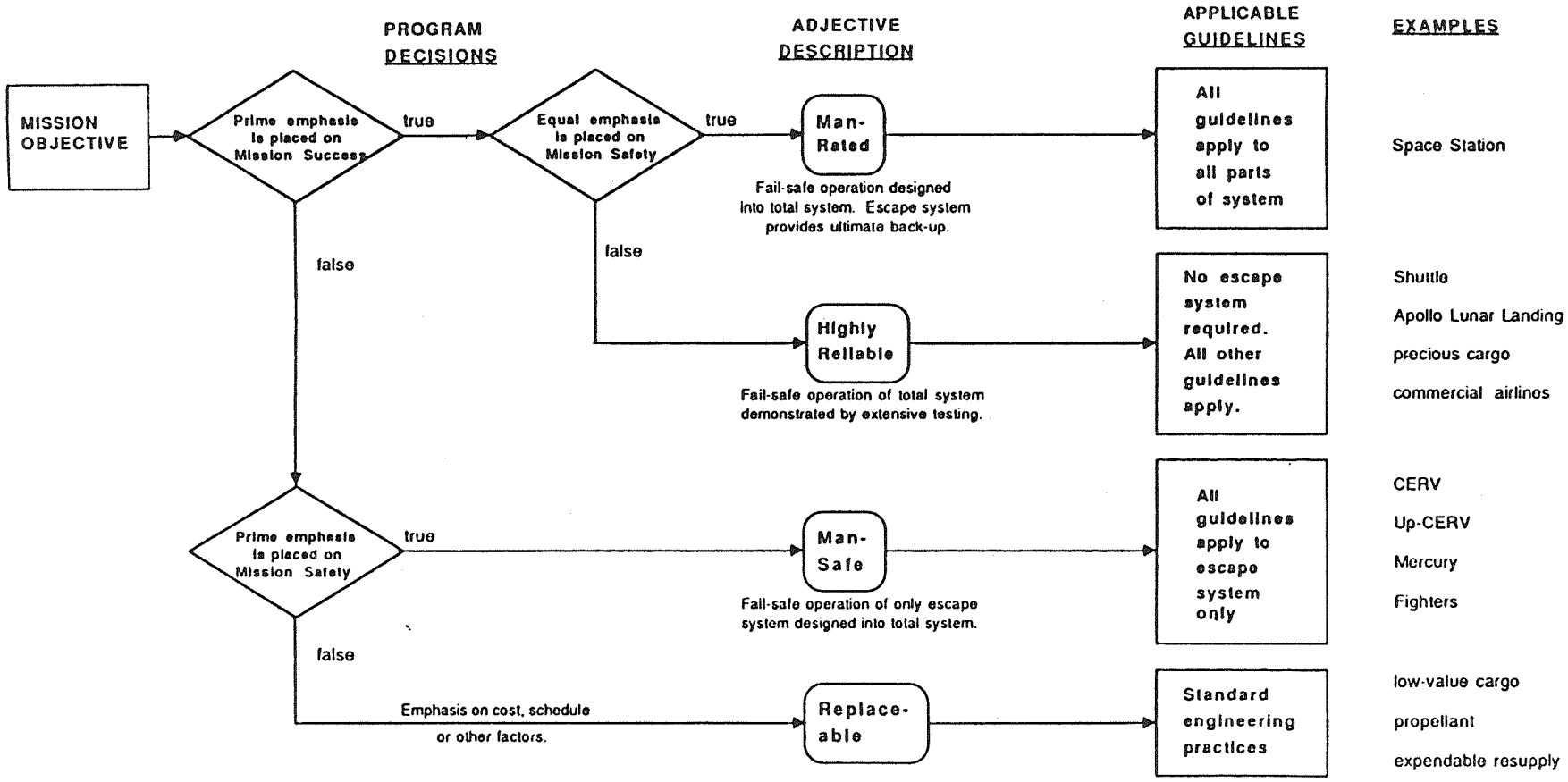


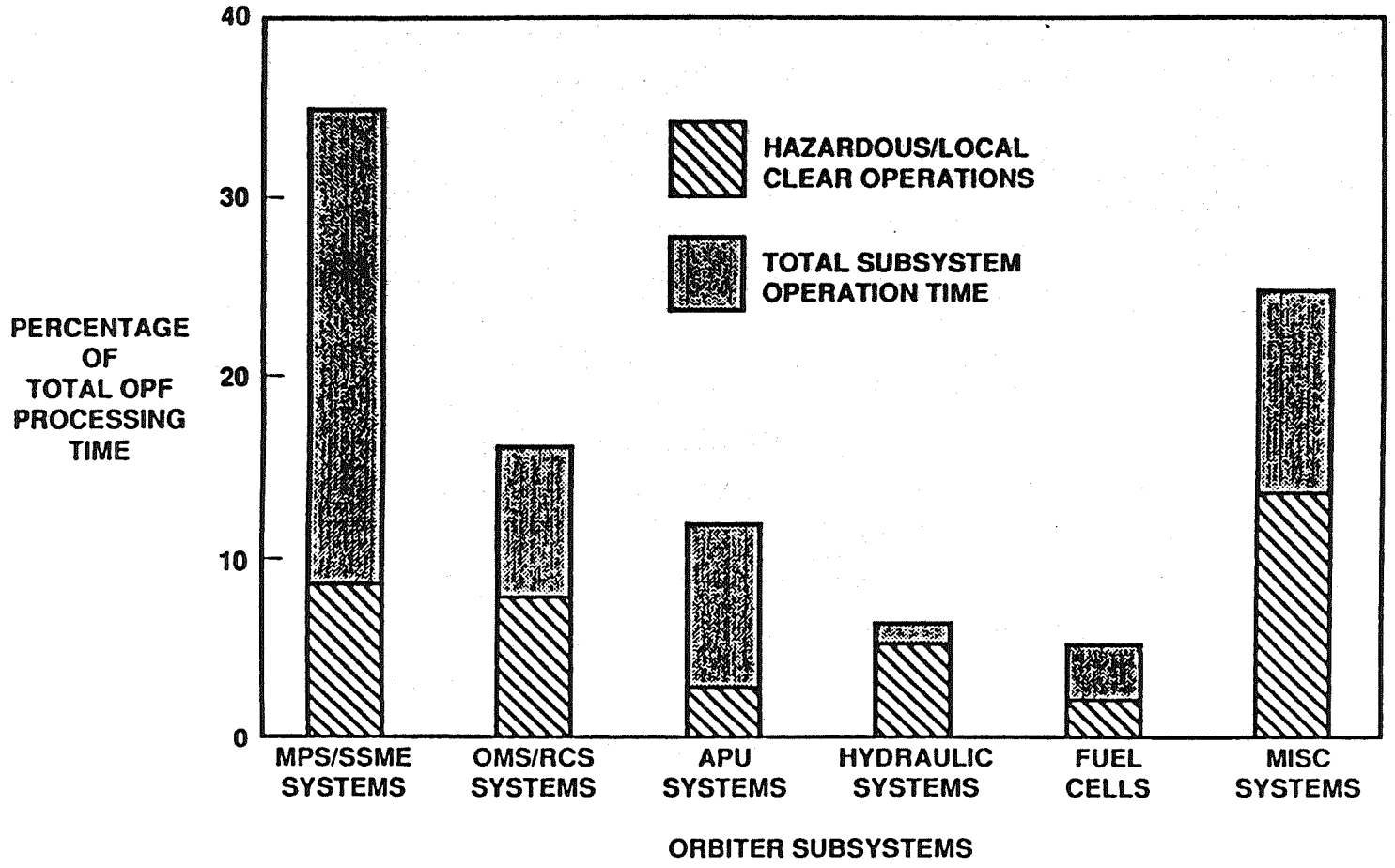
Figure 7

DESIGN GUIDELINES FOR MAN-RATING SPACE SYSTEMS

1) ENVIRONMENTAL CONDITIONS FOR OPERATIONS
2) CREW ESCAPE SYSTEM
3) FAILURE TOLERANCE : Primary structure, pressure vessels and thermal protection systems should be designed for zero tolerance for any failure or malfunction that would jeopardize crew safety.
 - Design of all other critical systems should ensure:
 a) No single failure results in a critical hazard.
 b) No two failures result in a catastrophic event.
4) HAZARD DETECTION AND SAFING
 - Vehicle Diagnostic Systems
 - Fire Suppression Capability
5) STRUCTURAL / MATERIALS CRITERIA
6) REDUNDANCY
 - Appropriate functional redundancy on all critical systems (i.e. fail-safe, fail-operational/fail-safe, etc.)
7) DISPLAYS AND CONTROLS
 - System status monitoring and failure alerts

Figure 8

ORBITER PROCESSING TIME BY SUBSYSTEM



SOURCE: Rockwell International

MPS/SSME SUBSYSTEMS REQUIRE MOST OF THE ATTENTION

Figure 9

PROPULSION ISSUES AND REQUIREMENTS

- o PROPELLANT CANDIDATES (Safety, Reliability, Cost, Commonality)
- o SYSTEMS APPROACH TO PROPULSION DESIGN
- o FAULT TOLERANCE
 - Design for Benign Failure Modes / Failure Containment (Eliminate gears, provide lubrication, etc.)
 - Engine-Out Capability
- o ONBOARD CHECK OUT AND VERIFICATION CAPABILITY
 - Propulsion System Verification at Operational Conditions
- o HEALTH MONITORING AND CONTROL
 - Accurate Hazard Detection and Real-time Diagnostic Capability
 - Highly Reliable/Redundant Systems
 - Technology Development Program
- o THROTTLING CAPABILITY
 - Needs To Address System Requirements (G-Limiting, Q-Control, Engine-Out)
 - High Reliabilty (Implications to Abort Capability)
- o COMPREHENSIVE TEST AND VERIFICATION PROGRAM
 - Establish True Propulsion Failure Limits (Off-Limits Testing and Test-to-Failure)
 - Verify Failure Modes and Effects
- o OPERATIONALLY EFFICIENT (Low Cost, High Reliability and Maintainability)
 - Design for Simplicity and Accessibility; Implement Nondestructive Evaluations
 - Quality Assurance

Figure 10

SHUTTLE DERIVATIVES - UNMANNED

AND

BOOSTER PROPULSION - LIQUIDS/HYBRIDS

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NEXT GENERATION EARTH-TO-ORBIT SPACE TRANSPORTATION SYSTEMS

UNMANNED VEHICLES

&

LIQUID/HYBRID BOOSTERS

Uwe Hueter

George C. Marshall Space Flight Center
Program Development
Space Transportation & Exploration Office

June 26, 1990

NEXT GENERATION EARTH-TO-ORBIT SPACE TRANSPORTATION SYSTEMS

UNMANNED VEHICLES & LIQUID/HYBRID BOOSTERS

by
Uwe Hueter

Abstract

The United States civil space effort when viewed from a launch vehicle perspective tends to categorize into the pre-Shuttle and Shuttle eras. The pre-Shuttle era consisted of expendable launch vehicles where we matured a broad set of capabilities in a range of vehicles, followed by a clear reluctance to build on and utilize those systems. The Shuttle era marked the beginning of the U.S. venture into reusable space launch vehicles and the consolidation of launch systems used to this one vehicle. This led to a tremendous capability, but utilized man on a few missions where it was not essential and compromised launch capability resiliency in the long term.

Launch vehicle failures, between the period of August 1985 and May 1986, of the Titan 34D, Shuttle Challenger and the Delta vehicles resulted in a reassessment of U.S. launch vehicle capability. The reassessment resulted in President Reagan issuing a new National Space Policy in 1988 calling for more coordination between federal agencies, broadening the launch capabilities and preparing for manned flight beyond the Earth into the solar system. As a result, the Department of Defense (DoD) and the National Aeronautics and Space Administration (NASA) are jointly assessing the requirements and needs for this nation's future transportation system. Reliability/safety, balanced fleet and resiliency are the cornerstone to the future.

This paper provides an insight into the current thinking in establishing future unmanned earth-to orbit (ETO) space transportation needs and capabilities. The paper presents a background of previous launch capabilities, future needs, current and proposed near term systems and system considerations to assure future mission needs will be met. The paper focuses on propulsion options associated with unmanned cargo vehicles and liquid booster required to assure future mission needs will be met.

Presented @ " Space Transportation Propulsion Symposium",
Pennsylvania State University, June 25-29, 1990.

NEXT GENERATION EARTH-TO-ORBIT SPACE TRANSPORTATION SYSTEMS

UNMANNED VEHICLES & LIQUID/HYBRID BOOSTERS

by
Uwe Hueter

Introduction

Effective space exploration requires reliable transportation, a balance of good science, and a progressively expanding space infrastructure starting with the Space Station. Adequate, reliable, lower cost space transportation is a key to the nation's future in space. Primary in the critical near term, is making more effective use of the systems we have and evolving a few early flexibility enhancements.

Launch vehicle failures, between the period of August 1985 and May 1986, of the Titan 34D, Shuttle Challenger and the Delta vehicles resulted in a reassessment of U.S. launch vehicle capability. Also, the country's total reliance on the Space Shuttle (SS) for all manned transportation and the majority of the unmanned satellites was questioned. The reassessment resulted in President Reagan issuing a new National Space Policy in early 1988, changing the nation's space transportation policy. The policy calls for more coordination between federal agencies, broadening the launch system base for assured access, and sets as a national goal manned flight beyond the Earth into the solar system. As a result, the Department of Defense (DoD) and the National Aeronautics and Space Administration (NASA) are jointly assessing the requirements and needs for this nation's future transportation system. Reliability, fleet balance and resiliency are cornerstone to the future.

The Space Shuttle will remain the primary manned access to space for many years and upgrades are planned to improve reliability, safety, and operational efficiencies. Key among these upgrades are: development of an Advanced Solid Rocket Motor (ASRM) to provide improved reliability through redesign and advanced manufacturing facilities; continuing design and process adjustments to our current solid rocket motor; and completing the Space Shuttle Main Engine (SSME) new high pressure turbopumps along with other select design improvements to address every critical failure mode. Other areas of key improvement are: upgraded state-of-the-art Orbiter subsystems such as avionics; additional crew escape capability; potential design improvements in the external tank; and launch/turnaround/flight operational changes to reduce cost per flight.

Acknowledgement for contributions to this paper is given to Mr. Tom Mobley from Martin Marietta Corporation and Dr. James Steincamp & Mr. David Taylor from NASA/MSFC.

In addition, added flexibility is needed in transportation systems by the late 1990s, including addition of: heavy lift capability complementary to the Space Shuttle to assure delivery of Space Station transportation node hardware, lunar and planetary vehicles, and other key payloads; an additional Orbiter to provide downtime for servicing and protect the fleet capability from significant mission disruptions; and an assured crew return vehicle (ACRV) for safe return of the crew from Space Station Freedom. Early requirements could be met by a vehicle such as Shuttle-C. Post year 2000 requirements will establish a need for a new unmanned modular, low cost launch vehicle such as the Advanced Launch System (ALS) and perhaps new liquid or hybrid rocket boosters for mission reliability, safety and flexibility. The exact timing of each needs focus, but certainly system understanding should mature and major steps need to continue in related technologies through our base technology and test bed efforts along with the directed technology initiatives planned by both the NASA and the AF.

It is clear that national space activities should take advantage of the many unfolding opportunities through a balanced science and infrastructure program. Transportation systems remain a vital enabling ingredient in accomplishing these objectives. It is time now to continue moving ahead on a course of continuity and challenge.

This paper provides an insight into the current thinking in establishing future unmanned earth-to orbit (ETO) space transportation needs and capabilities. The paper presents a background of previous launch capabilities, future needs, current and proposed near term systems and system considerations to assure future mission needs will be met. The paper focuses on propulsion options associated with unmanned cargo vehicles and liquid booster required to assure future mission needs will be met.

Lessons Learned

The launch vehicle failures of 1985-1986, brought into sharp focus that today's launchers fall far short of the kind of near-perfect reliability expected of space transportation vehicles. Figure 1 summarizes the experience of the world's major launch vehicles, past and present. The term "success ratio" rather than reliability highlights an important qualification to this tabulation: the number of launches of any one vehicle configuration is too small, from a statistical perspective, to yield an actuarially dependable reliability estimate. In particular, those vehicles with the largest number of launches have evolved from the ballistic missiles of forty years ago through both incremental and block upgrades. Moreover, the underlying data behind these summary results is an "apples and oranges" mixture, e.g. the expendable vehicle failures include some upper stage failures while the Shuttle data does not. Figure 2 depicts past launch rates. In recent years, the Soviets have been launching vehicles at a rate of approximately five times that of the rest of the world. Since the Soviets usually have one or two failures each year, there is at least ground for suspecting that reliability of current launch vehicles may approach a practical limit of approximately 0.98, i.e. one loss in every 50 launches. Figure 3 illustrates an intuitively expected trend of reliability growth with vehicle evolution: successive versions of the Titan vehicle more quickly achieved higher reliabilities than their predecessors. Similar trends have been calculated for other vehicles. Although the number of failures due to any one factor is small, there is some indication that the early failures are due primarily to redesign, while later failures relate to manufacturing and operational processes.

Launch Vehicle	Launches	Mission Failures	Success Ratio
Saturn	33	0	1.000
Atlas	495	94	.810
Titan III	152	7	.954
Delta	194	12	.938
Space Shuttle	35	1	.971
Ariane	27	5	.815
Proton (D-series)	165	18	.891

Figure 1. Launch Vehicle Success History

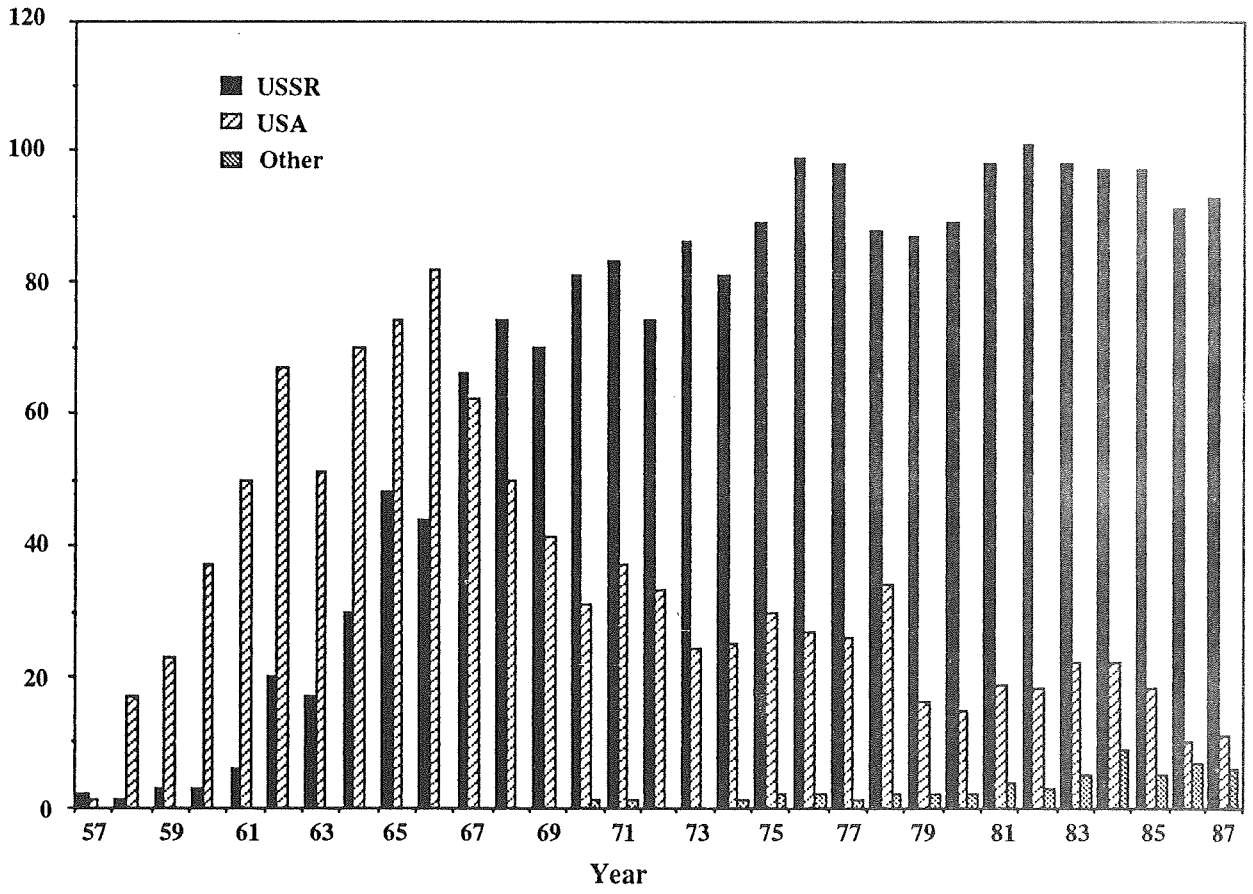


Figure 2. Annual Launches

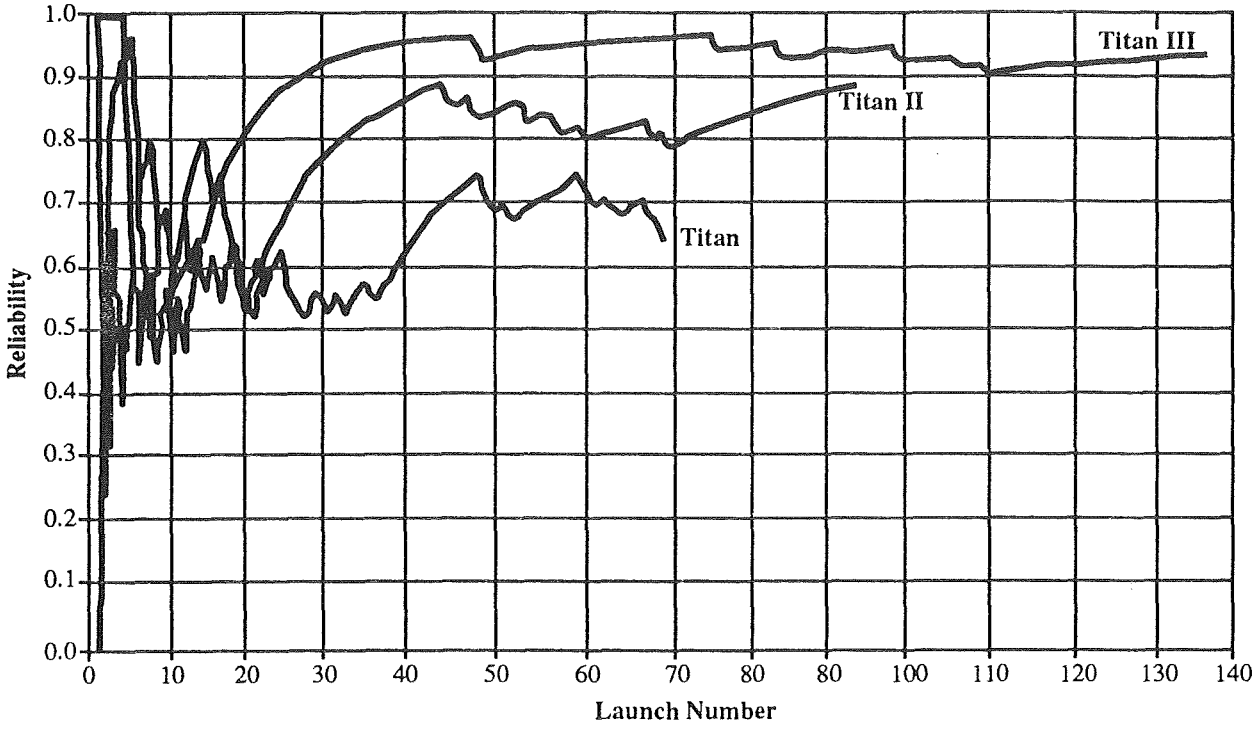


Figure 3. Reliability Estimation of Titan I, II, III

These trends have implications for the Shuttle. The initial run of 24 successful launches was a very respectable showing for a new vehicle, although longer success runs have been observed (e.g. 43 for the Delta). During the 32 month stand down following the Challenger accident, extensive improvements were made, not only in the solid boosters, but also throughout the vehicle and the supporting operational and management systems. Further, programs such as the advanced turbopump and the ASRM were undertaken or planned to further enhance the Shuttle's reliability. In the longer run, improvements which allow the SSMEs to operate at constant or reduced throttle will further improve reliability. In short, the evolution of the Shuttle has begun. Additionally, the Space Shuttle has unique advantages relative to current expendables, in the form of redundancy and abort modes which can in many situations save the crew, vehicle, and payload in the event of malfunctions. Nonetheless experience now tells us that achieving launch vehicle reliabilities greater than 0.98 is a challenge rather than an accomplishment.

The above discussion leads one to conclude that a goal to achieve a perfect launch vehicle is not a very pragmatic approach. Instead, a more reasonable approach that allows the nation to both plan and budget for eventual failures will prevent a repeat of the nation's stand down experienced after the Challenger accident. In addition, alternate vehicles to allow launching either manned or unmanned cargo would assure the nation a capability to continue operation even in the event of a catastrophic failure. Reassessment of U.S. space policy has resulted in the following objectives:

"• Assured access to space, sufficient to achieve all United States space goals, is a key element of National Space Policy
• U.S. space transportation systems must provide a balanced, robust, and flexible capability with sufficient resiliency to allow continued operations despite failures in a single system
• Goals of U.S. space transportation policy are:
 - Achieve and maintain safe and reliable access to, transportation in, and return from, space
 - Exploit the unique attributes of manned and unmanned launch and recovery systems
 - Encourage U.S. private sector space transportation capabilities without direct federal subsidy
 - Reduce costs of space transportation and related services"[1]

Budget /Cost Considerations

The estimated NASA budget requirements for the next ten years is shown in Figure 4. The budget includes the operating fund (R&PM), construction of facilities (COF), and program costs segregated into the major NASA's offices. The budget wedge for each office, except the Office of Space Flight (OSF), includes both approved programs and projected new starts. The OSF wedge only includes approved programs. The heavy dark line indicates a NASA budget growth of 15% through 1993 and 5% in 1994 and beyond. OSF potential new starts include such programs as Shuttle-C, liquid rocket booster (LRB), Space Transportation main/booster engines, space transfer vehicle (STV) and an assured crew return vehicle (ACRV). As can be seen from Figure 4, zero budget would be available to institute new initiatives if the budget growth rate is limited to 15%. This emphasizes the need for

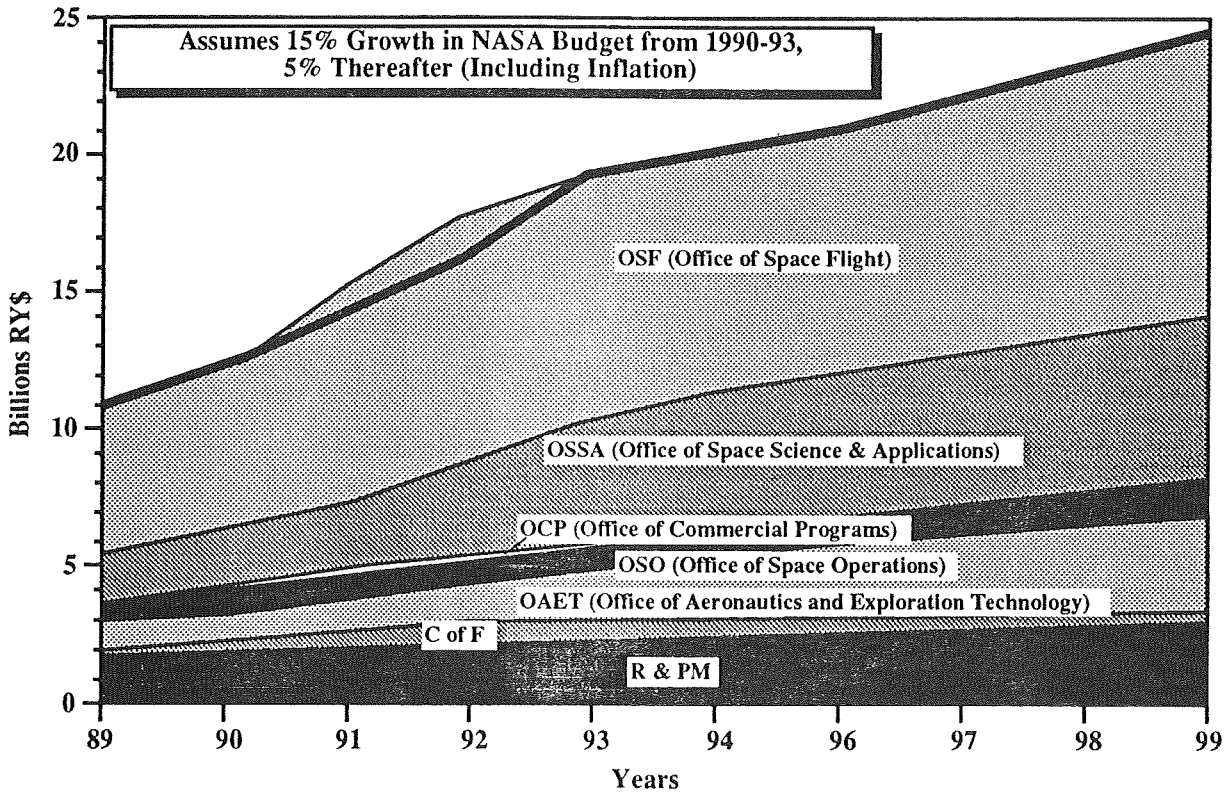
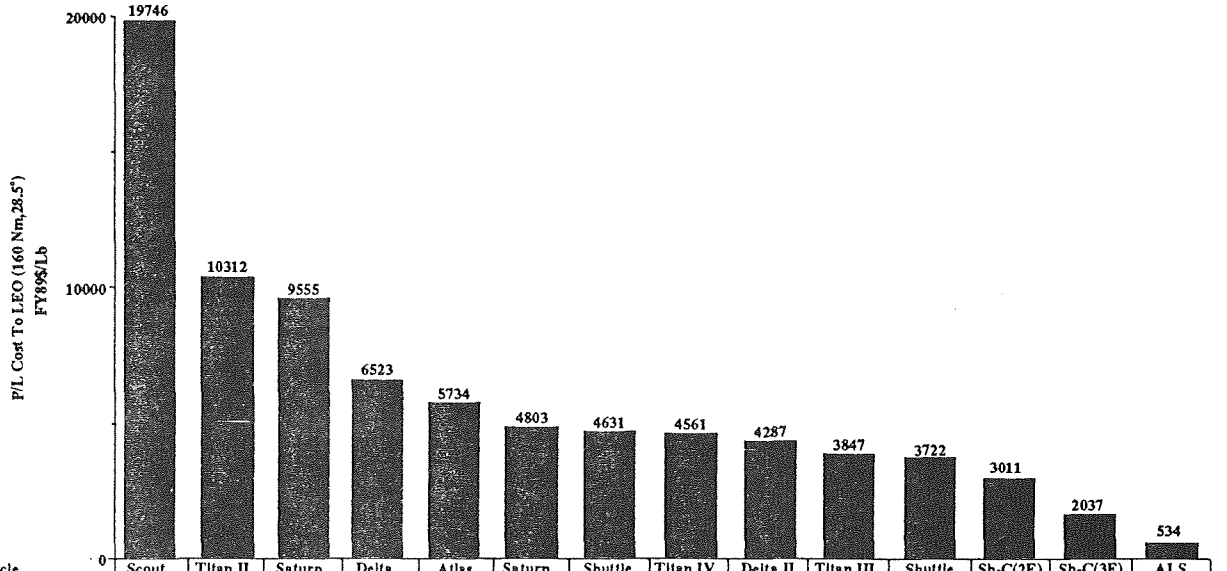


Figure 4. Space Transportation Planning Budget Wedge Analysis

[1] "Space Flight", Office of Space Flight - National Aeronautics and Space Administration, March 1989.

reducing current program recurring cost to allow a budget wedge for new initiatives. Additionally, the new starts will require low upfront investments to remain within the budget. Toward this end, investigations are currently being conducted to improve operational efficiencies of current launch systems. The average cost per flight and dollars per pound of payloads to LEO for various launch vehicles (past, present and future) are shown in Figure 5. Each bar represents each vehicle's expected matured flight rate per year. All costs shown reflect expendable hardware and operations costs and exclude the vehicle's design, development, test and engineering (DDT&E) and reusable hardware costs. As can be seen from the figure, dramatic cost reductions are anticipated for future launch vehicles.



Vehicle	Scout	Titan II	Saturn IB	Delta (3920)	Atlas Centaur	Saturn V	Shuttle Baseline	Titan IV	Delta II	Titan III	Shuttle ASRM	Sh-C(2E)	Sh-C(3E)	ALS OBJ
P/L To LEO (Lbs) (160 nm, 28.50)	500	5000	31000	7400	13200	249000	45000	38000	8700	28800	56000	89000	147300	150000
Assumed Flt Rate	4	5	2	4	4	2	14	10	12	5	14	3	3	20
Avg Cost Per Flt (89$/M)	10	52	296	48	76	1196	208	173	37	111	208	268	300	80

Note: Above Estimates Include Expendable Hardware And Operations (Exclude DDT&E And Reusable Hardware)
Both Fixed And Variable Operational Costs Are Included
Shuttle-C Estimate Includes Shuttle-C Unique Fixed Costs And Variable Cost And Excludes Common Shuttle
Fixed Cost (i.e. Marginal Cost). Based On Alternating Flights Of 3 Engine And 2 Engine Configurations Respectively.

Figure 5. Launch Vehicle Operations Cost Estimates

The current Space Shuttle cost per flight and the projected reduction is shown in Figure 6. The breakout of the projected cost per flight for the Space Shuttle is shown in Figure 7. As can be noted from the figure, operations cost constitute a major percentage (43.9%) of the total projected cost per flight. This figure is based on a flight rate of 14 flights per year.

The question of reusable versus expendable launch systems is always a major cost consideration in the initial phases of a new design. The argument has been that reusable launch vehicles, although higher in DDT&E cost, are more cost effective than expendable vehicles based on the longer term. To support the argument of reusability, Figure 8 provides a cost comparison for the projected cost of the Space Shuttle SRBs. Included in the cost is the refurbishment cost for the SRBs. As can be seen, a cost savings of approximately $56 million can be achieved by recovering rather than expending the boosters. Thus, the trend of future vehicles will be to recover the major elements of the system.

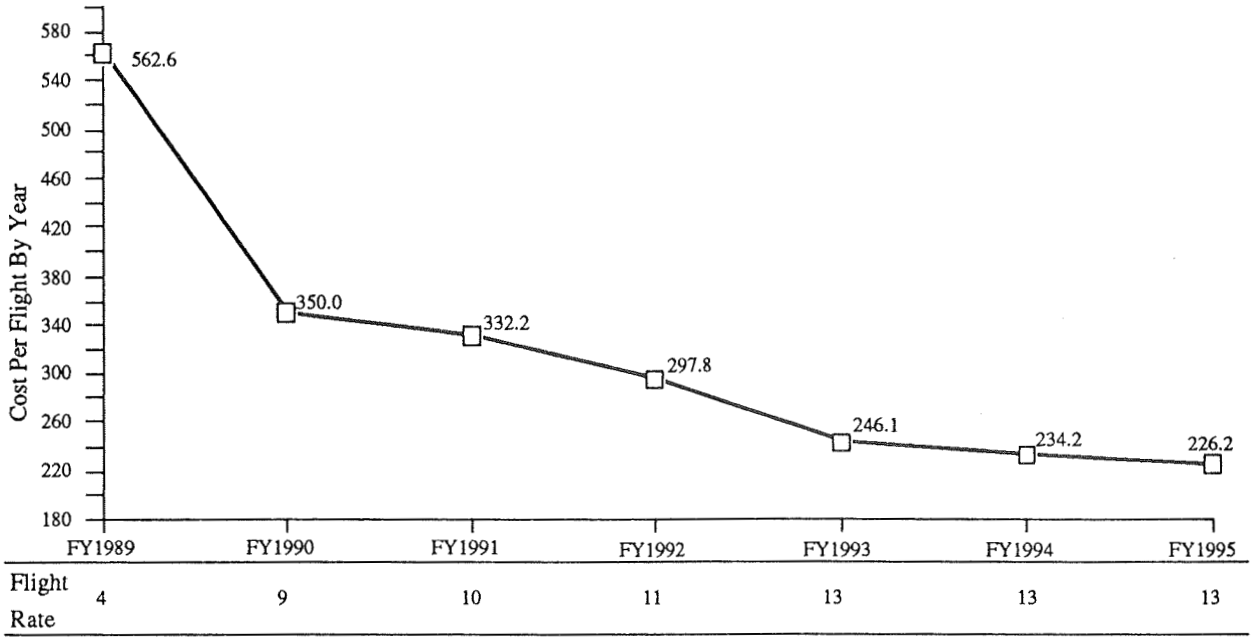


Figure 6. Space Shuttle Cost Per Flight Projection (Millions of FY 1989 Dollars)

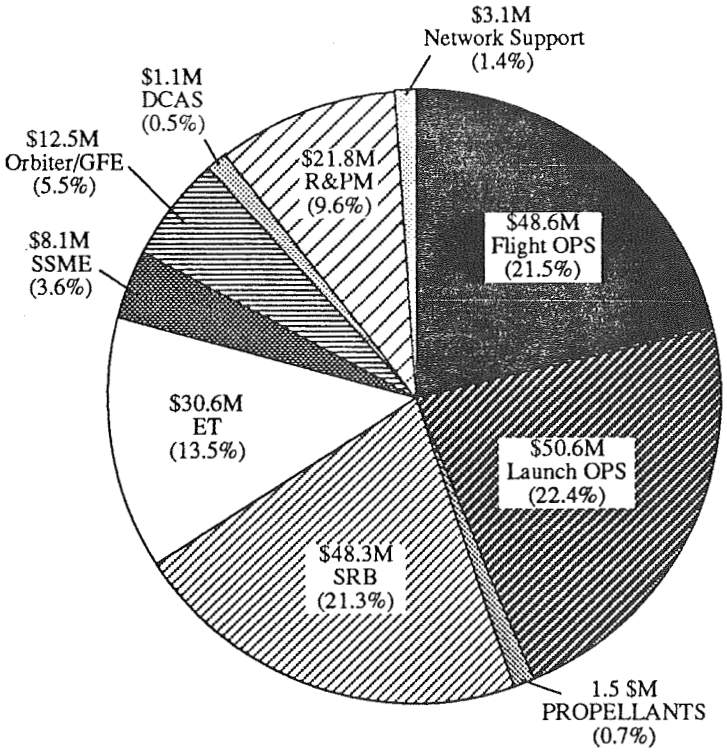


Figure 7. Space Shuttle Cost Per Flight (Millions of FY 1989 Dollars)
$226M/Flight in 1995

	Recovered/Refurbished Per Flight Set (2 Units)	New Per Flight Set (2 Units)	DELTA Cost
SRM	$28,390,000	$47,490,000	$19,100,000
BAC	$11,860,000	$49,760,000	$37,900,000
KSC Costs	$1,000,000	$0	($1,000,000)
Total	$41,250,000	$97,250,000	$56,000,000

Figure 8. SRB Cost: Mid 90's (Millions of FY 1989 Dollars)

The budget environment, along with NASA's current needs, indicates that the potential for new starts will be severely limited for many years. The few new starts that will be approved for NASA will most likely require non-optimum (stretched) development schedules to reduce near term funding profiles.

Early And Long Term Mission Needs

The Civil Needs Data Base (CNDB) is a projection of the civil space transportation requirements for the time interval 1990 thru 2010. The current version of the CNDB is referenced as CNDB '90. There are presently two options included in the CNDB '90, the base mission model and the expanded mission model which includes the Space Exploration Initiative (SEI). Figure 9 graphically shows the projected range of mass required to be launched into low earth orbit. In terms of total mass

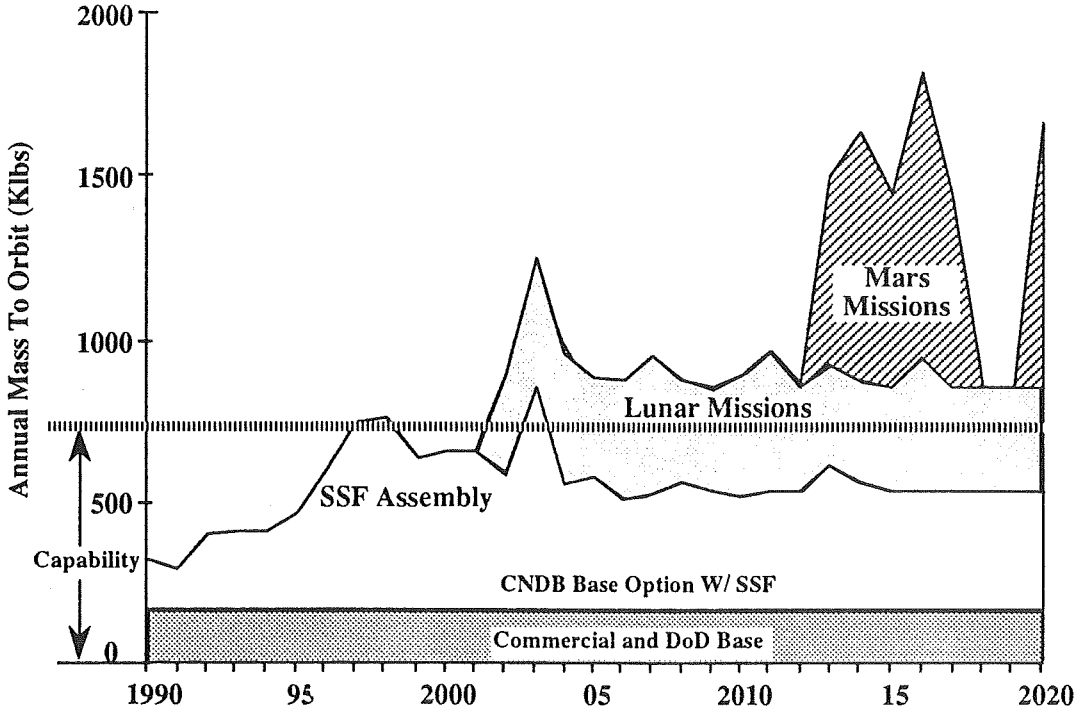


Figure 9. Space Transportation Requirements Forecast

required to be delivered to low earth orbit, the SEI missions (Mars and Lunar) are clearly the most demanding. As noted on Figure 9, additional launch capability will be required. Some of the infrastructure requirements for SEI missions are:

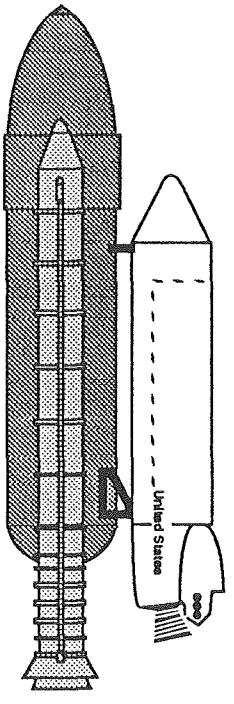
- A heavy lift launch vehicle (HLLV)
- Earth orbit facilities for assembly and support
- Mars & Lunar transfer systems
- Science payloads and equipment

Future Systems Studies

Unmanned Vehicles

Future unmanned transportation systems such as Shuttle derived vehicles (SDV), sidemounted and inline cargo carriers, and Advanced Launch Systems (ALS) are being studied. The Lunar/Mars missions definitely will require an HLLV to maintain the flight rates and orbital assembly to a minimum.

NASA is currently analyzing various SDV evolution paths to establish the desired direction for future unmanned launch vehicles. One SDV heavy lift concept currently being studied for the late 1990's is the Shuttle-C, see figure 10. The Shuttle-C is a largely expendable, unmanned launch system capable of carrying payloads of 85,000-150,000 pounds to low earth orbit. Shuttle-C is not a new system, but rather an expansion of our current Space Shuttle Program. It uses existing and modified



- **Standard 4-Segment SRB's (Reuseable)**
- **Standard ET (Expendable)**
- **Orbiter Boattail (Expendable)**
 - 2 SSME's (Remove SSME #1)
 - Remove Verticle Stabilizer
 - Remove Body Flap
 - Cap SSME #1 Feedlines
 - OMS Pods (Do Not Install OME's, RCS Tanks And 4 RCS Thrusters/Pod)
 - RCS Performs Circularization And Deorbit
 - Cover And Thermally Protect SSME #1 Opening
- **Payload Carrier (Expendable)**
 - New Shroud/Strongback
 - Skin/Stringer/Ringframe Construction Of Al 2219
 - 15' X 82' Useable Payload Space
 - 15' X 60' Changeout On Pad Capability
- **Avionics**
 - Uses Mature Design Components From STS And Other Applications
 - Requires Some New Integration And Software
- **Performance - ETR** - **160 NM/28.5° - 114 Klb**
 - **220 NM/28.5° - 109 Klb**

Figure 10. Shuttle-C (Cargo)

Shuttle qualified systems and the established Space Shuttle infrastructure to achieve the earliest possible heavy-lift capability, as well as other benefits of economy and reliability. The major new element that is required is the Shuttle-C Cargo Element (SCE). Some design and definition work is needed to develop the SCE, but it is a relatively straight forward concept. A key aspect is that it is designed to allow payloads to be interchangeable with the Orbiter. The SCE structure is built in two major elements. The forward payload carrier is an easily manufactured aluminum skin-and-ring frame fuselage. Payload bay length is 82 feet and is covered by Orbiter-like doors. The aft (boattail) fuselage is based on existing Orbiter design, minus wings, vertical stabilizer, and body flap. Although some aspects of Shuttle-C are being refined, the design is well understood. The SRBs and ET are identical to those in the inventory, which reduces costs and minimize disruptions in the Space Shuttle program. The Main Propulsion System (MPS) is also identical to the current Orbiter MPS. Two SSMEs are used for payloads up to 100,000 pounds to low Earth orbit, with three used for payloads in excess of 100,000 pounds. SSME's used by Shuttle-C will have seen as many as nine missions on the Orbiters and will complete their life cycle on Shuttle-C. The on-orbit propulsion is provided by an aft reaction control system (RCS) based on the Orbiter design. The Orbiter's maneuvering engines are not needed, and the remaining thrusters will be configured to meet Shuttle-C RCS requirements. The payload environment will be equal to that of the Orbiter with simpler, low-cost systems replacing the expensive, reusable Orbiter systems. Avionics/GN&C are adapted from the Orbiter; those systems required for manned support, long-duration orbit, descent, and landing will be deleted. Other SDV options being studied are inline vehicles utilizing both ET derived or SRB replacement sized liquid boosters, hybrid boosters, and recoverable propulsion/avionics modules. Potential evolution paths of the various SDV options are shown in Figures 11 and 12.

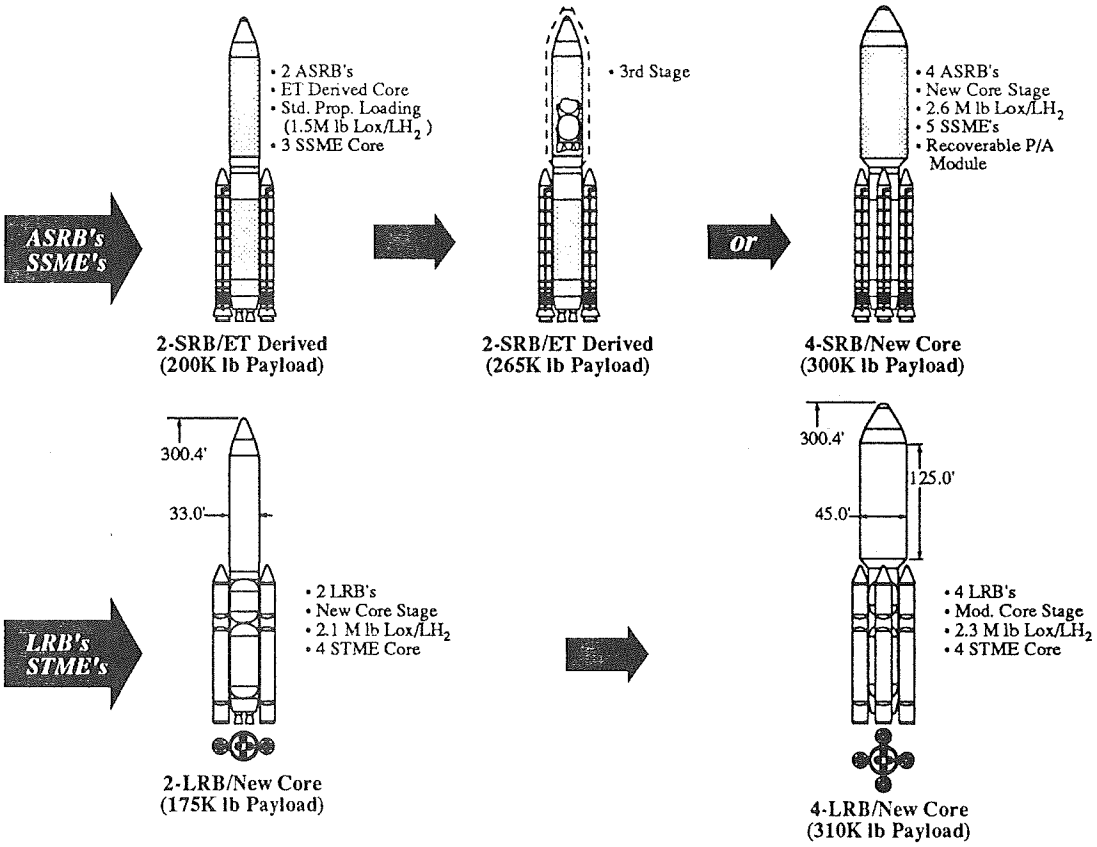


Figure 11. Potential Shuttle Derived Evolution

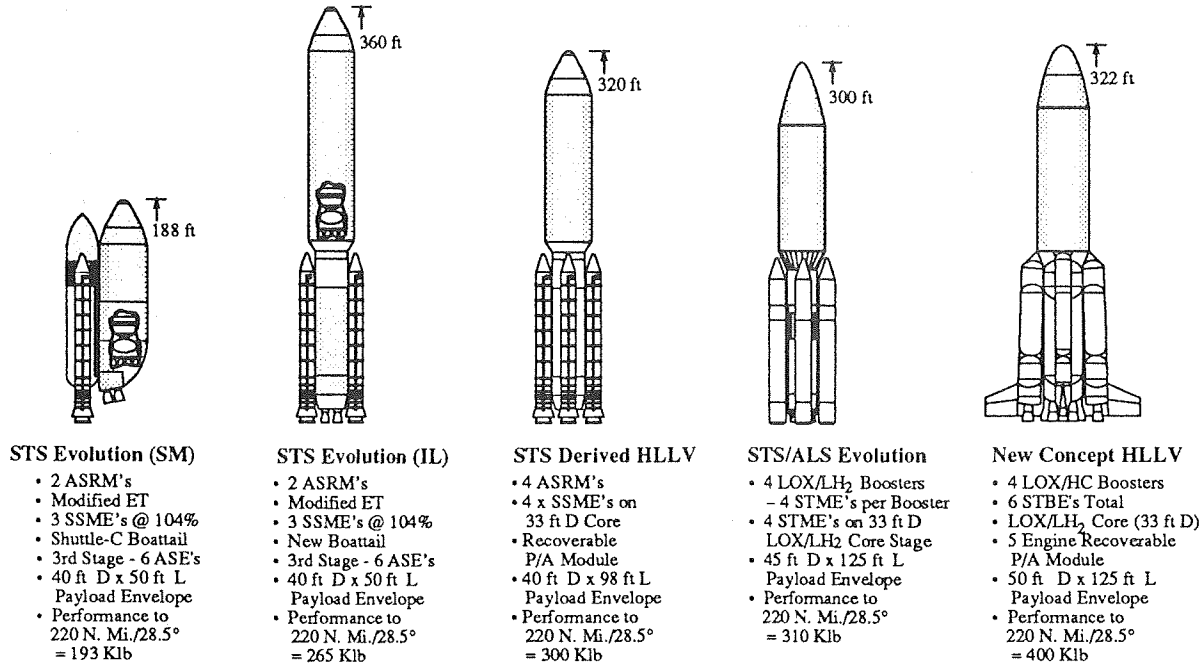


Figure 12. Shuttle Derived Heavy Lift Launch Vehicles

The ALS program, a joint AF/NASA effort, is conducting both studies and advanced development activities to determine a family of unmanned vehicles required to meet future mission needs. The range of payload lift capability to LEO being investigated is from approximately 40,000 to 450,000 pounds, see Figure 13. NASA has a lead role for ALS in liquid engine systems and technology. The goal of ALS is to provide a low cost unmanned payload lift capability in the range of $300 per pound to LEO.

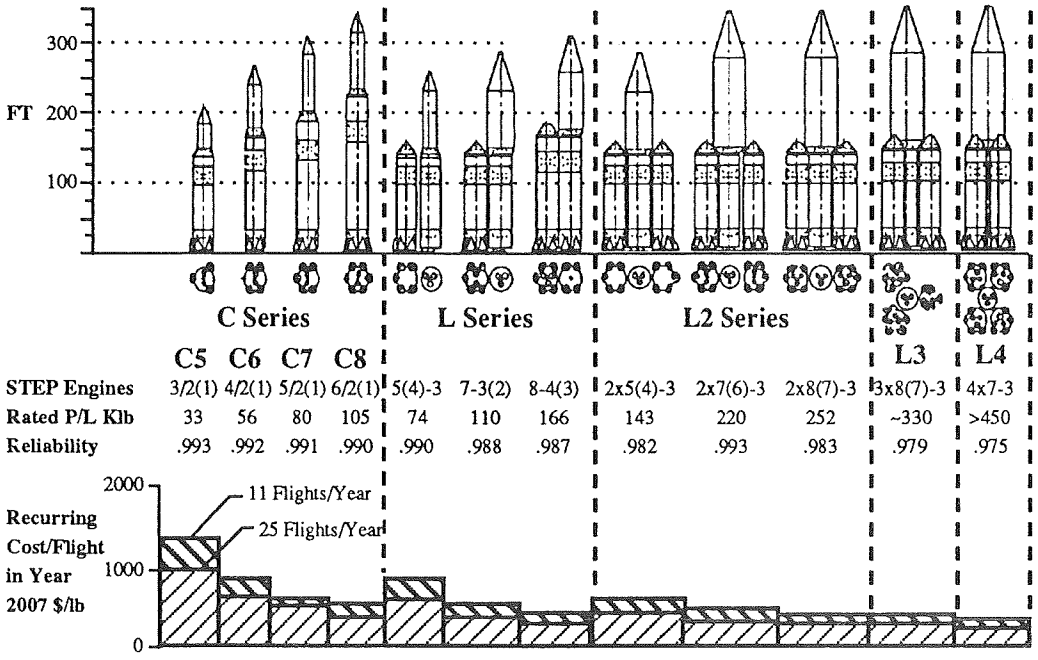


Figure 13. Advanced Launch System Family

The SEI 90-Day Study resulted in both an SDV and ALS option for satisfying the requirements for the Lunar and Mars missions, see Figures 14 & 15 respectively. The primary focus of the 90-Day Study was to provide a low DDT&E cost approach for implementing SEI. Therefore, Shuttle derived vehicles utilizing existing and growth elements were proposed. The alternative was a low operational cost philosophy for which ALS was chosen. Current efforts are underway to study alternate infrastructure approaches for satisfying the integrated ETO requirements, including both manned and unmanned launch vehicles.

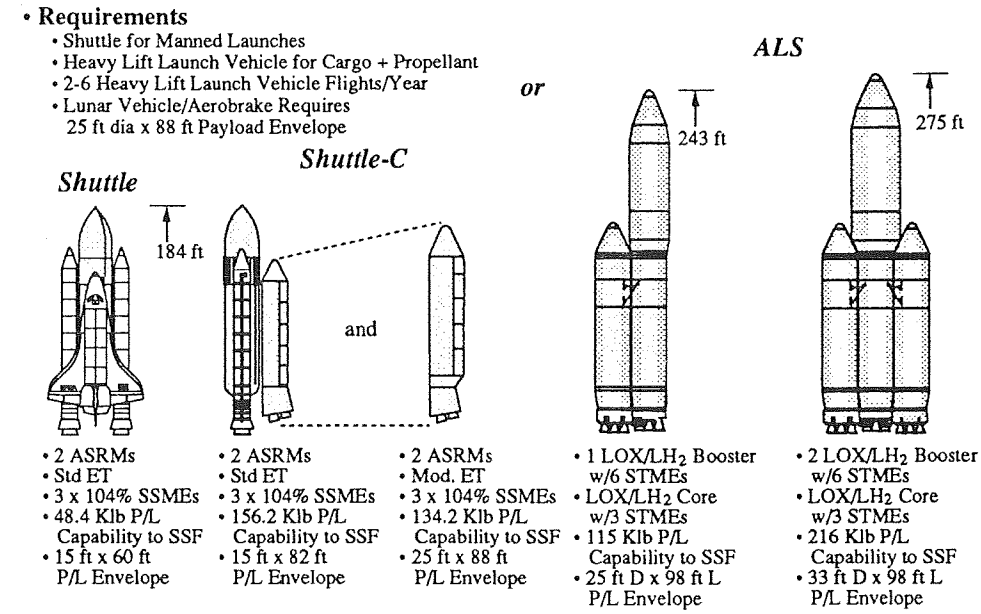


Figure 14. Launch Vehicles for Lunar Missions

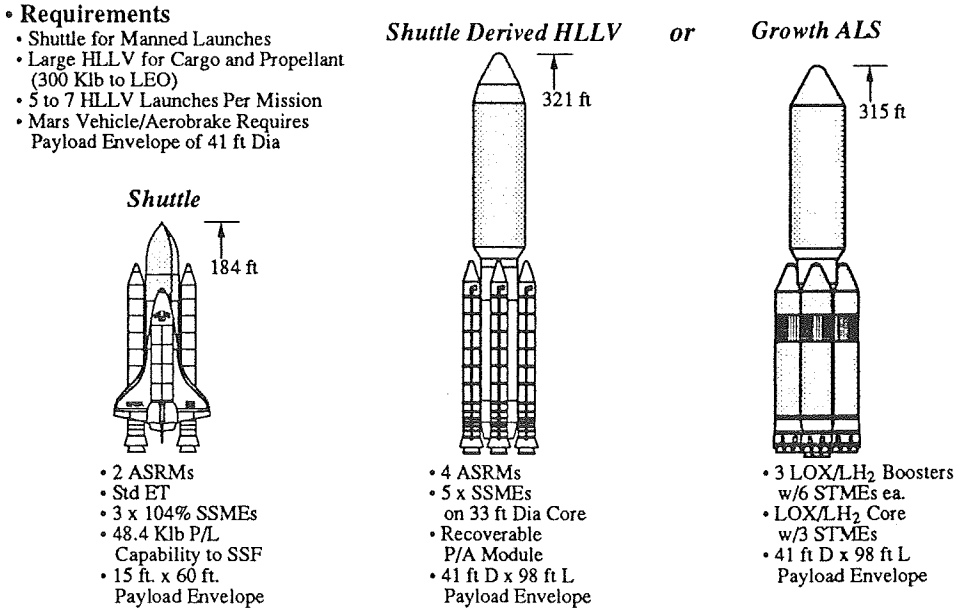


Figure 15. Launch Vehicles for Mars Missions

As previously stated, the nation's need for access to space is expected to grow significantly during the next 15-30 years. Highly efficient and flexible space transportation systems will be needed to support a number of new space initiatives currently in the planning phase. These space transportation systems will range from the current space Shuttle with planned improvements to heavy-lift launch vehicles using new booster propulsion systems operating on liquid oxygen/liquid hydrogen, liquid oxygen/hydrocarbon, and liquid oxygen/solid fuel propellants.

The National Aeronautics and Space Administration (NASA) is expected to be in the forefront of these developments and will be called upon to provide the needed technology and development. Therefore, it is imperative that NASA foster, nurture, and continue to develop its capability in the full spectrum of rocket propulsion. Potential propulsion options must be continuously explored and assessed to ensure that the most optimum systems for the particular applications are understood and characterized. In order to accomplish this, technology programs with a specific focus must be initiated in sufficient time to provide the detailed knowledge needed to make the proper selections.

Booster Options

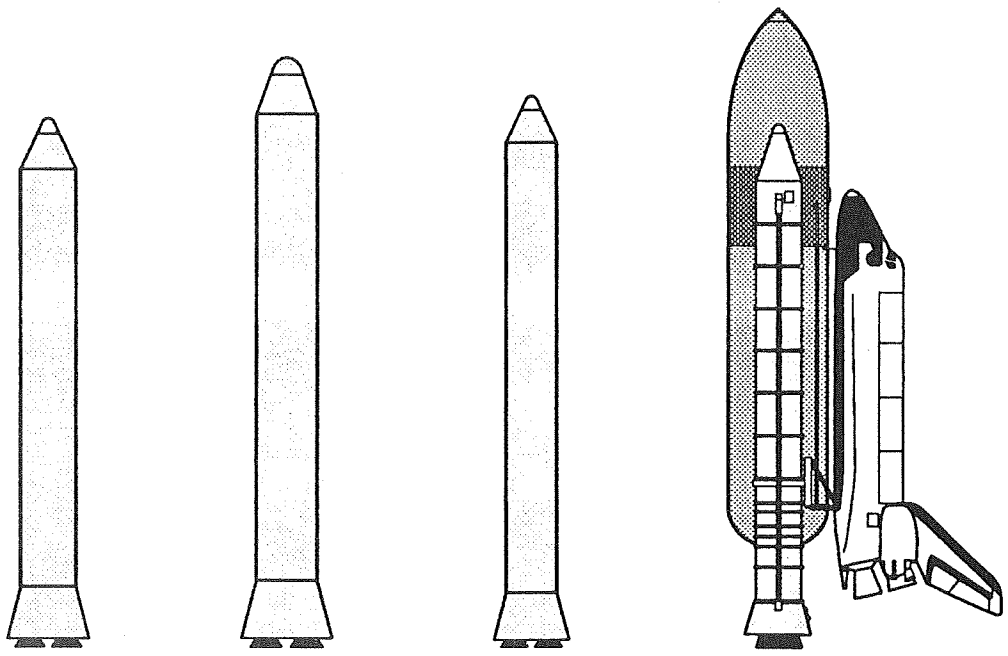
Solid Rocket Boosters have a significant flight performance database. The simplicity of their propulsion system design results in low cost and high reliability. The high propellant density of solid boosters results in the smallest system packaging for any given thrust level. This reduced envelope minimizes the booster structural cost and launch site processing facility requirements. The significant drawback of the solid rockets on the Shuttle is that no abort options are available after booster ignition and prior to motor shut down. The inability to shut down a solid motor on command precludes any first stage abort modes. In addition to limiting mission abort options, the SRB also produces combustion products which significantly impact the environment. The SRB and planned advanced SRB motor exhaust contains significant amounts of hydrochloric acid (HCl) and aluminum oxide. The HCl contributes to the acid rain problem and is suspected of reducing the ozone layer in the atmosphere. The aluminum oxide is suspected of contributing to Alzheimer disease. Because the oxidizer and the fuel are mixed and loaded in the motor cases at remote propellant manufacturing locations, special safety precautions have to be taken during SRB handling, shipping, and assembly prior to installation on the Shuttle vehicle. Extensive safety requirements increase operational costs and timeline schedules. For example, the SRB stacking activities at the vehicle assembly building (VAB) require that the building be evacuated of all unnecessary personnel during these assembly sequences.

Therefore, studies and technology activities are ongoing to provide the database and technology maturity to allow either liquid and/or hybrid boosters to be designed and built when needed. The primary study focus to date has been on boosters to replace the solids on the Shuttle. The following discussion deals primarily with boosters of that class. However, larger liquid boosters are being investigated for application to a heavy lift launch capability. The technologies described are also applicable to this class of boosters.

Liquid Rocket Boosters

While liquid rocket boosters (LRB) offer increased mission safety because they provide engine out capability and thrust termination on command, the liquid propulsion systems are more complex and costly compared to the SRBs. The unit cost estimates for the liquid booster options range from 15 to 30 percent higher than the solid boosters.

There are several LRB propulsion system options, see Figure 16. Each option has advantages and disadvantages compared to the others, and their rating of merit in various criteria, i.e. cost, reliability, etc., fluctuates such that no clear choice is available. The following paragraphs describe the more promising liquid booster propulsion system options for the Space Shuttle and summarize their pros and cons. It should be noted that the LRBs described have the performance to deliver a 70,500 lb payload to 28.5° inclination and 160 nautical miles with 75 percent engine power level (engine out capability). This greatly exceeds the SRB or proposed ASRM capability. A comparable SRB would require a motor casing diameter increase to fourteen feet.



	Pressure Fed LO_2 /RP-1	Pump Fed LO_2 /LH_2	Pump Fed LO_2 /RP-1	SRB
Length	163 ft.	178 ft.	151 ft.	149 ft.
Diameter	16 ft.	18 ft.	15 ft.	12 ft.

Figure 16. Liquid Rocket Booster Configurations

LO_2/LH_2 Pump-Fed LRB - The LO_2/LH_2 LRB is the largest Shuttle booster option because of the low density of the hydrogen fuel. This vehicle is approximately 18 ft in diameter and 178 ft high. Because of its size, the LO_2/LH_2 booster presents the most Shuttle integration difficulties. The booster does have the advantage of common propellants with the Shuttle Main Propulsion System (MPS); and if the Space Shuttle Main Engines (SSME) were replaced with Space Transportation Main Engines (STME), the booster and MPS would have common propellants and common engines.

Current costs predictions for the liquid booster options do not show an advantage for any vehicle. However, the LO_2/LH_2 booster costs are based on the STME technology goal of $3.5M per engine. Escalated STME engine costs could require that the engines be recovered for reuse. The added system complexity and higher propulsion system costs would put the LO_2/LH_2 option at a disadvantage.

Another consideration for the pump-fed LRB options is the inherent reduced reliability for turbo machinery. STME design and operating parameters are intended to maximize the total system reliability and should result in a minimum of criticality one failure modes.

LO_2/RP-1 Pump-Fed LRB - The LO_2/RP-1 booster is the smallest selected liquid booster option for the Space Shuttle. The booster length is 151 ft and the diameter is approximately 15 ft. This booster presents the minimum Space Shuttle integration impacts of the selected LRB options. The propulsion system design is very conservative and operates at combustion chamber pressure (Pc) comparable to the F-1 engine used on the Saturn 1C launch vehicle. More optimum LO_2/RP-1 engines are a consideration which would significantly reduce the booster size. The current operational Energia LO_2/RP-1 booster engine operates at a Pc three times the proposed Shuttle LO_2/RP-1 system. This higher Isp propulsion system would make the pump-fed LO_2/RP-1 booster comparable in size to the solid booster systems.

The overall reliability of the LO_2/RP- 1 pump-fed system would still be influenced by the use of turbo machinery. The low Pc engines would have an advantage because of lower pump requirements, but require larger propellant supplies. The higher pump requirements for the more efficient LO_2/RP-1 engines would present similar reliability issues to the high pressure LO_2/LH_2 systems.

LO_2/RP-1 Pressure-Fed LRB - The pressure-fed LRB has the highest reliability propulsion system because it does not utilize turbopumps to produce the high pressure propellants which are injected into the combustion chamber. The use of high pressure (1000 psi) propellant tanks instead of pumps results in thick (1 inch) tank walls and therefore is the heaviest of the three liquid booster options. The low engine combustion chamber pressure (660 psi) also requires the highest propellant mass of the three options. However, the high density of the RP- 1 compared to LO_2results in a lower total propellant volume than the LO_2/LH_2 booster. The pressure-fed LO_2/RP-1 LRB is 16 ft in diameter and 163 ft long.

The pressure-fed LRB requires some technology demonstration unique to this propulsion system cycle. Although many pressure-fed systems have been flown successfully, e.g. the Shuttle orbital maneuvering system, these systems are relatively small compared to the LRB. The development of systems to pressurize large propellant tanks has yet to be achieved. A second propulsion system issue is the performance of large, low Pc thrust chambers especially with high range (40%) throttle capability. These technology issues are being investigated by the Booster Technology Program at MSFC. Demonstration test articles are being designed and developed for both the pressurization system and thrust chambers. The quarter scale system testing is scheduled to be completed by 1993.

Although the pressure-fed booster costs are only slightly lower than the pump-fed systems, the pressure-fed system does not have the high cost risk associated with the $3.5M pump-fed engines. Cost proposal for full scale (750K thrust, Pc=660 psi) pressure-fed thrust chamber assembly test articles support the current cost estimates for the production of pressure-fed engines ($2.5M each). Any escalation of the pump-fed engine costs give the pressure-fed boosters a significant advantage over the pump-fed options.

Hybrid Rocket Boosters

The Phase I Hybrid Booster Technology Study was completed by four aerospace contractor teams. The study teams recommended booster options which used either a classical hybrid combustion cycle or a gas generator hybrid combustion cycle. The classical hybrid contains no oxidizer in the solid propellant grain and introduces liquid oxygen at the front end of the hybrid motor. The gas generator (GG) hybrid has a low percentage of oxidizer in the solid grain. When the GG is ignited, a fuel rich gas is produced in the motor and forced into an aft mounted combustion chamber. Liquid oxygen is injected into the aft combustion chamber to complete the fuel combustion.

The preliminary data developed in the Phase I study does not show a performance or cost advantage for either hybrid option. The vehicle size and costs are comparable to the SRB or ASRB. The discriminators between the two hybrid options are: (l) combustion cycle complexity and operating pressures; (2) manufacturing, transportation, and handling considerations; and (3) technology requirements.

Classical Hybrid Rocket Booster - The classical hybrid booster uses no oxidizer in the solid fuel. The hybrid motor is inert and presents no extraordinary manufacturing, handling, or transportation safety concerns. In addition, the combustion products of the classical hybrid are comparable to a hydrocarbon liquid fuel. The classical hybrid motor operates at approximately 1000 psia and would have motor casing design and manufacturing similar to solid rocket motor casings. Because no solid fuel and oxidizer mixing is involved in loading the motor cases, a monolithic case design case can be readily achieved for any size classical hybrid motor. The operating pressure of a classical hybrid can be achieved by either a pump or pressure-fed oxidizer system.

The key technology associated with the classical hybrid, see Figure 17, is the ability to inject liquid oxygen into the motor such that uniform combustion exists along the length of the solid fuel grain. Multiple port designs in the solid grain appear to be a promising solution, but very little testing on large motors has been accomplished to date. Ignition of the classical hybrid requires uniform oxidizer flow throughout all ports of the solid grain period. As the number of ports increase, the uniform ignition and burning throughout the fuel becomes more complex. The inability to provide uniform combustion in the motor would impact motor performance and result in numerous technical and safety issues.

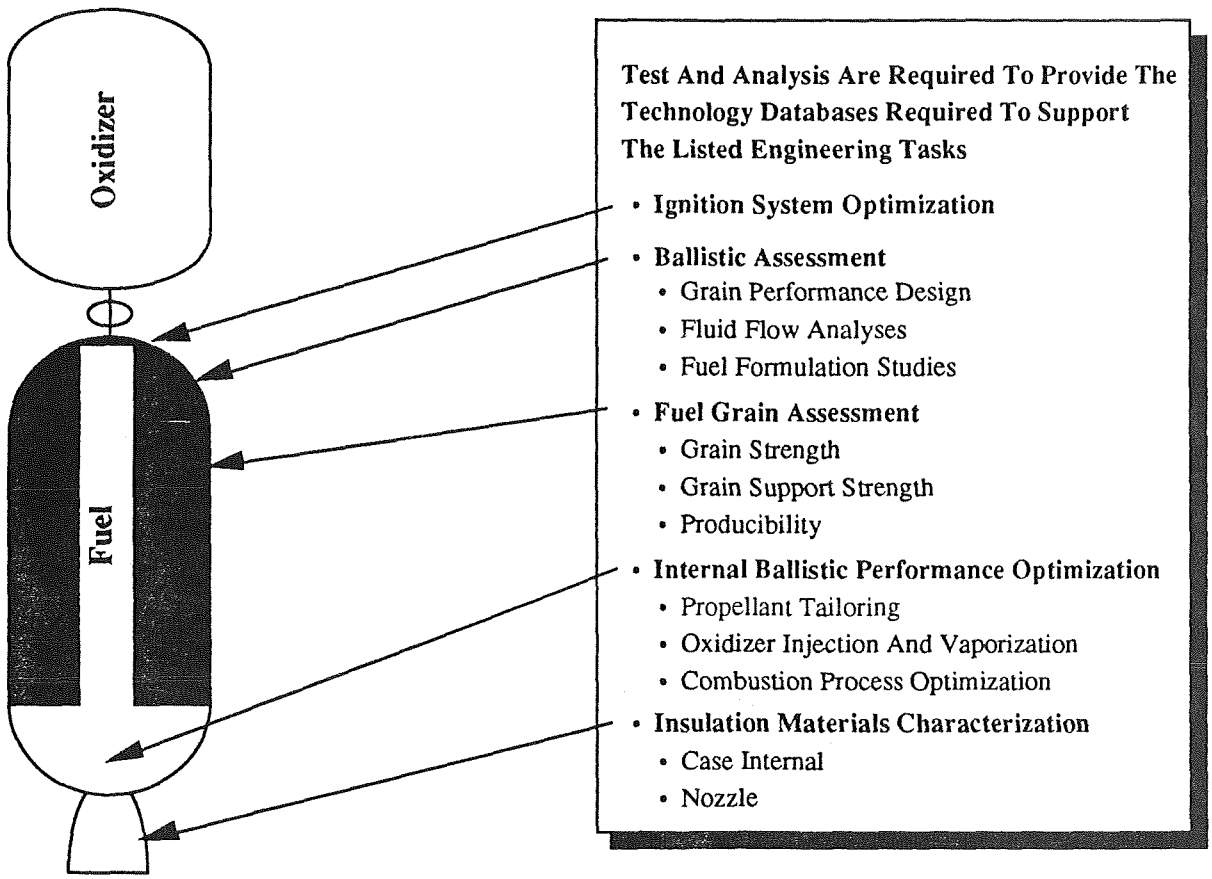


Figure 17. New Hybrid Technology Requirements

Gas Generator Hybrid Rocket Booster - The gas generator hybrid motor avoids the concern with uniform motor combustion by including a low percentage of oxidizer in the solid grain and injecting the liquid oxygen into the fuel rich solid motor combustion gas in a liquid rocket type combustion chamber. This combustion cycle minimizes the hybrid motor technology and relies on finely tuned liquid rocket combustion technology to provide safe, uniform, solid fuel combustion.

Data from the Phase I hybrid studies showed gas generator pressures from 1400 psi to 1870 psi. The corresponding aft combustion chamber pressures are 1000 psi to 1700 psi. The percent of oxidizer in the solid grain for all GG concepts is approximately 20% by weight. It is important to note that LO_2 engine inlet pressures significantly in excess of 1000 psi would exclude the pressure-fed liquid

oxygen option from the gas generator hybrid booster. At high operating pressures, pressurization system size and complexity combined with structural mass of the oxygen tank would negate the reliability advantages of the pressure-fed system.

Low percentages oxidizer in the solid grain significantly reduce the safety concerns in solid propellant manufacturing, loading, and motor transportation and handling. However, some increased safety requirements should be expected when compared to an inert motor. The low percentage of solid oxidizer also allows the use of chemical scavengers to reduce the amount of HCl in the motor exhaust to an acceptable level. Although scavengers reduce the performance of the solid fuel, the requirement for environmentally safe combustion products will dictate their use.

As stated above, the key technology issues for the gas generator hybrid booster parallel liquid rocket combustion technology. The balance between gas generator operating pressure and aft combustion chamber pressure is critical to the safe and efficient combustion of the fuel rich gas developed in the solid motor. Well documented pressure fluctuations exist in solid motors which will greatly influence the liquid/gas combustion chamber stability requirements. The capability of the liquid oxygen pressurization (pump or pressure) control system, efficiency of the LO_2 injector, and combustion stability of the thrust chamber are key technical issues in the development of a large gas generator cycle hybrid rocket booster.

Ignition of the GG hybrid booster also presents several technical challenges. The gas generator hybrid must be ignited over a large portion of the exposed surface while the oxidizer is introduced into the aft combustion chamber. In order to have a predictable start, the two events must occur simultaneously. The thrust chamber combustion will choke the flow at the throat and communicate a back pressure to the solid grain to prevent self extinguishment. If at that time the majority of the surface of the solid grain is not ignited, the grain will not perform as intended.

Technology Programs

The primary technology programs at the MSFC relating to future transportation systems are the solid rocket motor integrity program, the liquid engine test bed, ALS technologies and the Civil Space Technology Initiative (CSTI).

The main focus of the solid rocket motor integrity program is on improvement of solid rocket motor reliability. Issues being addressed are test and verification procedures, analytical model data bases, experimental test for data, systems approach for improving reliability, process control measures and instrumentation/diagnostic capability. Specific areas being actively worked are propellants and insulation, nozzles, bond lines, combustion dynamics and integrity/verification techniques. This program has been underway since 1984, and is expected to continue through at least 1993.

The liquid engine test bed program provides off-line propulsion component and development type tests in a highly realistic cryogenic engine environment. For example, a new turbopump design can be added to an SSME test bed engine and evaluated for selected technology improvements. Specific areas of technology being addressed are combustion testing, large scale turbomachinery validation and health monitoring. The turbomachinery effort includes air and water simulation testing of flow

models as well as computational fluid dynamics analyses. The health monitoring effort is particularly active in measuring engine performance and sensing actual engine operating conditions.

ALS technologies being pursued are the advanced engine program, advanced avionics program, recovery system development, composite structure development, advanced manufacturing processes and base heating analysis. The advanced engine program, the largest effort, focuses on the next generation liquid rocket engine needs and characteristics. Cryogenic hydrogen is the primary fuel being considered. A significant emphasis is on achieving a low cost engine.

A Propulsion R&T Program has been initiated that covers the specific technology needs required for the development of a pressure-fed, LO_2/RP- 1 propulsion system. Provisions for the research and development of a liquid oxidizer/solid fuel hybrid propulsion system are also included. The focus of this program is partially driven by a recognized deficiency in the technological development of pressure-fed and hybrid booster systems. This program is not only needed to correct this deficiency, but also to revitalize the nation's space program involvement in advancing rocket propulsion technology, which has languished since the Space Shuttle became operational. It will also provide much needed engineering experience to individuals replacing retired personnel who where the pathfinders in the Apollo and early Space Shuttle design efforts.

The MSFC's CSTI effort is focused in three areas: Earth-to-orbit propulsion, booster technology and the aero-assist flight experiment (AFE). The earth-to -orbit propulsion program addresses analytical models for engine environments and component life; bearing, seal and turbine blade technologies; instrumentation for engine environments; engineering testing to validate models; and component/ test bed testing. The booster technology focuses primarily on the hybrid and the pressure-fed propulsion systems.

Summary

The advent of Space Station Freedom and future anticipated Lunar and/or Mars manned explorations, the nation will require both additional heavy lift capability for unmanned payloads and enhanced capability in the manned vehicle area. More reliable and less costly transportation will be the driving force for whatever vehicles this country will decide to place into service to support its needs.

Development of technology, supporting of science and building of a sound space transportation infrastructure is cornerstone to U.S. space leadership. This nation is back on track with its launch vehicles. However, we have a far way to go to realize our plans for the future. Based on the data presented in this paper, the following major points can be made:

- The U.S. has had an extremely successful space program to date.
- Reliance on a single vehicle for transportation to orbit is unacceptable.
- Launch vehicles will never be 100% reliable, therefore one has to program and budget for eventual failure.
- The current budget environment will not allow for multiple major new starts, therefore one has to build as much as possible on existing systems.

- Major reductions in current systems' recurring costs will be required to allow new starts and maintain the funding within anticipated budget allocations.
- Future systems, both unmanned and manned, are being studied.
- A heavy lift launch capability will be required to support SEI mission requirements.
- Liquid/hybrid boosters provide an attractive alternative to solid boosters.
- Continued technology work in advanced low cost engines, pressurization systems, and hybrid combustion processes is needed to assure an adequate data base for future system implementation.

BOOSTER PROPULSION - SOLIDS

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National Aeronautics and
Space Administration

June 26, 1990

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

Solid Rocket Propulsion

Briefing To:
Space Transportation Propulsion Symposium

Ronald L. Nichols
Mgr, Solid Propulsion Research and Technology Office (ER41)
FTS 824-2681
(205) 544-2681

Presented by C. Clinton, MSFC

Solid Rocket Propulsion

NASA'S Commitment to SRM use

- Planned use well into 21st Century
- Typically launch about 300 SRM's over 5 year period
- Approximately $30B of hardware-depend on successful SRM operation during 5 year period
- Historical success rate has proven to be about 98%

Improvements Needed

- Success rates must be improved for manned flight and high-tech hardware launches
- Costs must be controlled to remain competitive

Solid Rocket Propulsion

Shortfalls

Cultural

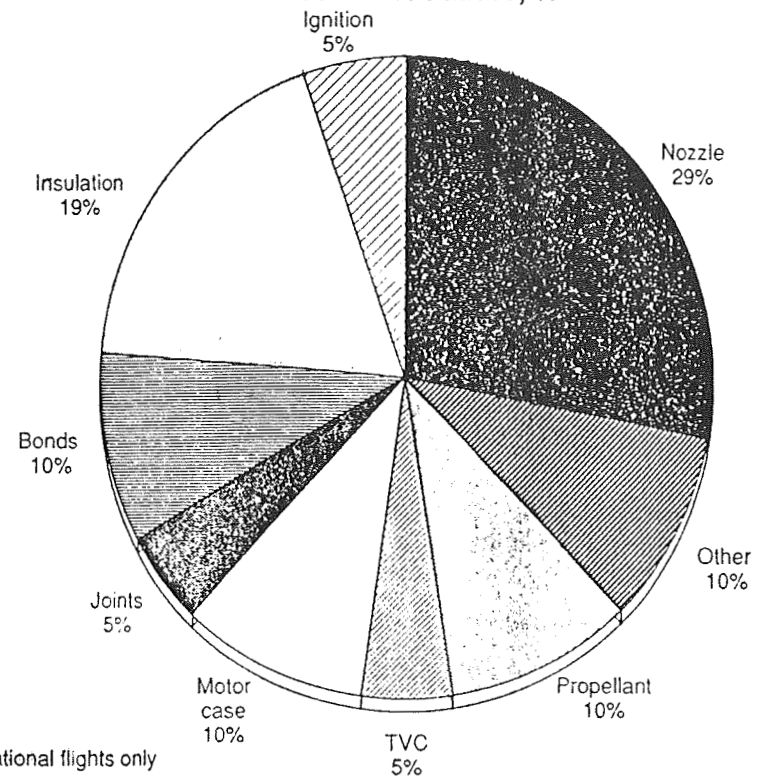
- Based on empirical approach – hot firings to prove success vs. technical understanding
- Extensive assumptions used in invalidated analytical models
- Designs based on tactical and strategic systems where 98% success rate is adequate
- Lack of fundamental understanding of engineering principals for design and analysis, processing and verification

Managerial/Leadership

- Absence of focused, continuous, coordinated government commitment and leadership during past two decades
- Major IR&D efforts in ballistics areas

Solid Rocket Motor Failure Database

Cause of Failure, %



Note: Operational flights only

Solid Rocket Propulsion

Current Programs

Solid Propulsion Integrity Program (SPIP)

Improve the success rate of the Nation's Solid Rocket Motors through

- Development of engineering base
- Generation of fundamental technical understanding of current SRM Technologies
- Providing tools for design, margin of safety prediction, process control, inspection and performance validation
- Controlled product variability with process sensitivity knowledge

Solid Rocket Propulsion

Current Programs (Continued)

- Redesigned Solid Rocket Motor Enhancements
 - Facilitization
 - Contamination Control
- Advanced Solid Rocket Motor Development
 - Specific components and system
 - Improved materials
 - Production automation

Solid Rocket Propulsion

Current Programs (Continued)

ALS/Low Cost Case, Insulation And Nozzle (LOCCIN)

- Attaching High Cost of SRM's
 - Innovative Designs
 - Low Cost Materials
 - Reduced Manufacturing/Fabrication Labor
 - Efficient Assembly/Checkout
 - Competition
 - Track Materials and Manufacturing Cost Savings
- Improving Reliability Through
 - Robust Designs
 - Verify Safety Margins
 - Define and Demonstrate Materials and Process Sensitivities
 - Set Materials and Process Specifications Based on Sound Accept/Reject Criteria
- Technical Maturity Achieved By
 - Laboratory Development
 - Sub-Scale Demonstration
 - Provide Technology for Full Scale Development

Solid Rocket Propulsion

Current Programs

Solid Propulsion Integrity Program – Engineering Base

	Predictive Capability	Materials Properties & Performance	Test Techniques	Process Understanding	Process Monitoring & Control	Improved Materials & Process	Instrumentation Development	NDE & Effects Of Defects
Composite Cases								
Propellant	X	X	X					
Nozzles	X	X	X	X	X	X	X	X
Bondlines	X	X	X	X	X		X	X
Internal/Flow	X		X				X	
Joints & Seals								
Integration & Verification	X	X	X	X		X	X	X

Solid Rocket Propulsion

Summary

- Progress being made in
 - Cultural
 - Managerial
 - Engineering base development
- Commitment is continuous through 1990's
- New initiatives that reduce cost and enhance reliability needed
- Solutions to environmental and flight safety issues should be aggressively pursued

Solid Rocket Propulsion

Additional Critical Issues/Recommendations

- Expand SPIP to complete matrix
- NASA involvement in clean propellant
- Develop thrust termination/restart SRM capability

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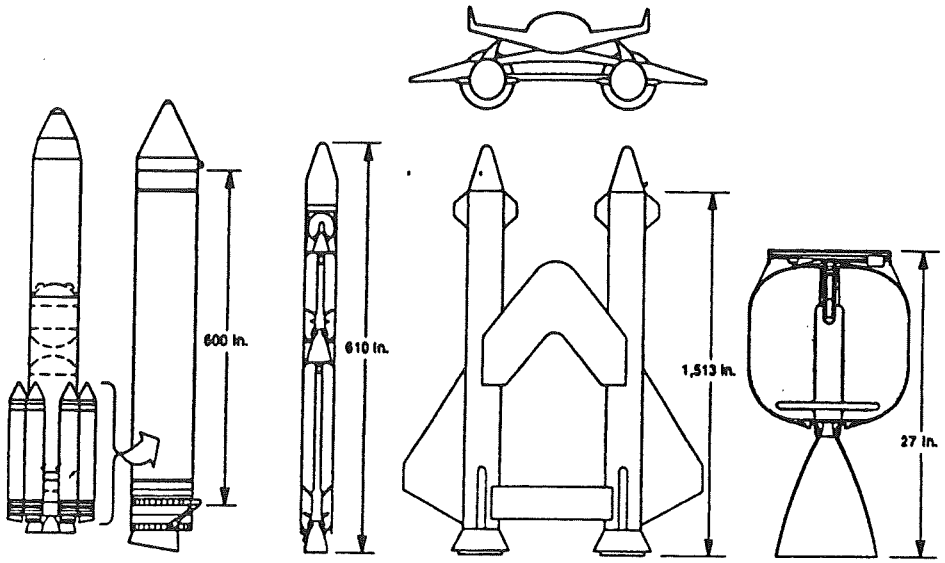
Next Generation Solid Boosters

R. K. Lund

27 June 1990

Thiokol CORPORATION

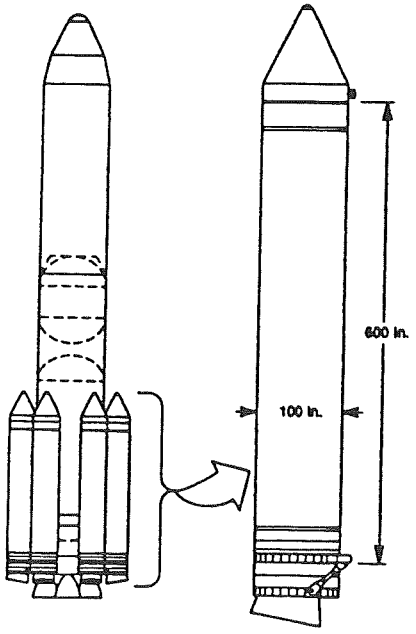
Space Transportation Solid Rocket Motor Systems



Large Launch Booster | Small Launch Vehicle | Reusable Flyback Booster System | Space Propulsion Motor

Thiokol CORPORATION

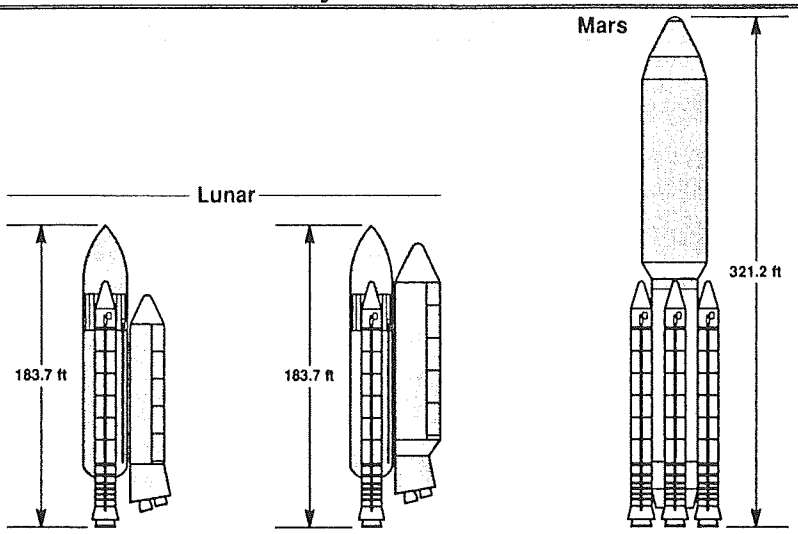
Large Launch Solid Rocket Boosters



- Concept objectives:
 - Reduce booster costs to $5–6/lbm of booster weight (60% decrease)
 - Increase booster reliability and safety (demonstrate 0.999X reliability/booster)
 - Clean propellant exhaust (no HCl)

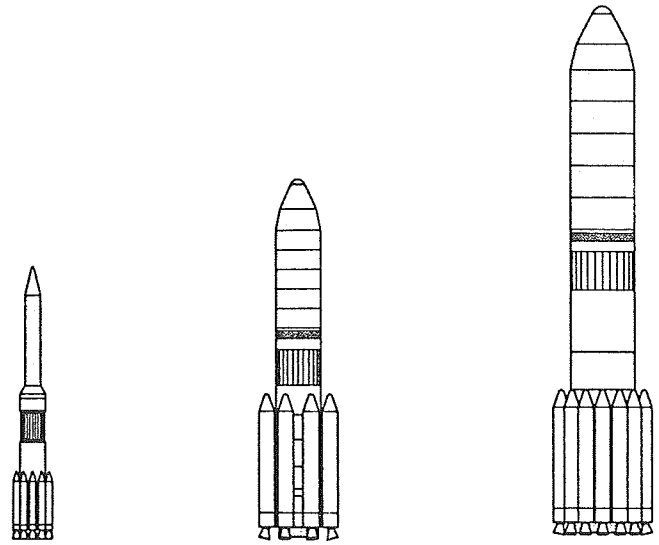
INFORMATION ON THESE PAGES WAS PREPARED TO SUPPORT AN ORAL PRESENTATION AND CANNOT BE CONSIDERED COMPLETE WITHOUT THE ORAL DISCUSSION

Shuttle–Derived Heavy Lift Launch Vehicles



Net Payload	71 t	61 t	140 t
Booster	2 ASRBs	2 ASRBs	4 ASRBs
Core Stage	Standard ET	Standard ET	32.8 ft dia
Core Propulsion	3 SSMEs	3 SSMEs	Recoverable P/A With 5 SSMEs
Payload Envelope	15.1 ft dia/ 82 ft length	24.9 ft dia/ 89.9 ft length	41 ft dia/ 98.4 ft length

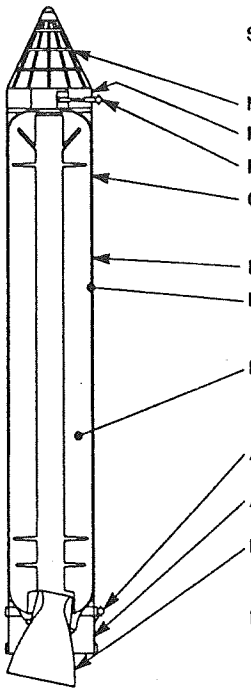
ALS–Derived Heavy Lift Launch Vehicles



Booster Thrust (klb)	8–460	6–600	12–600
Payload (klb)	88	117	250

Thiokol CORPORATION

Enabling Technologies



	Design	Process/Material
System	Reliability	S.A.F.E.Rsm
	Cost	Performance/cost optimization
Nose cone	Stiffened shell	Composite/aluminum
Forward skirt extension	Stiffened shell	Aluminum welded
Forward attach structure	Pivot	Midstrength D6AC
Case/skirts	Monolithic Integral aft dome Symmetric aft opening	Filament wound on "green" insulation Graphite prepreg resin
External insulation		Bonded cork
Internal insulation	Variable thickness Single material Flapless	Kevlar EPDM Strip wrap on mandrel Co–cure with case
Propellant/grain	Slotted CP	Clean exhaust Machined grain Batch or continuous mix Direct bond
Aft attach structure	Aft end thrust reaction truss structure	Midstrength D6AC
Aft skirt extension	Stiffened shell	Aluminum welded or chem–milled
Nozzle	Submerged centerline Canted boss Symmetric nozzle	PAN carbon phenolic Molded
Ignition/ordnance	Pyrogen/laser–initiated igniter Forward dome termination	Tailored clean propellant Shaped charge detach

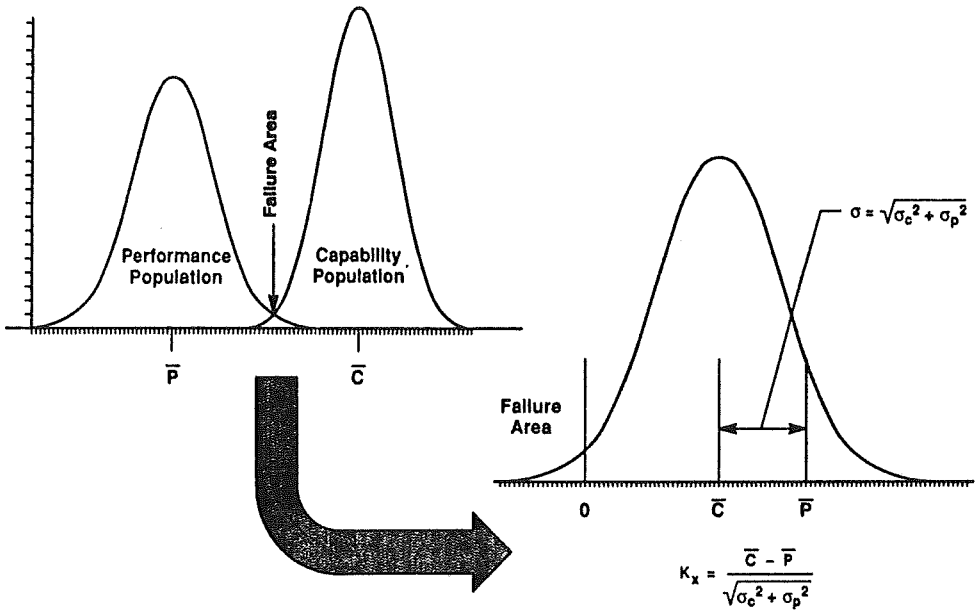
S.A.F.E.Rsm Philosophy

Statistical Analysis for Engineering Reliability

- Link reliability and producibility to affect design
- Conduct design to meet allocated reliability
 - Estimate design reliability based on estimated performance and capability distributions
 - Base capability distribution on historical test data and established requirements
 - Develop approach to estimate performance distribution from standard engineering models
- Link process control variables and key design variables to critical failure modes
- Establish test program to demonstrate reliability (tailor test data to establish capability and performance distributions)

Thiokol CORPORATION

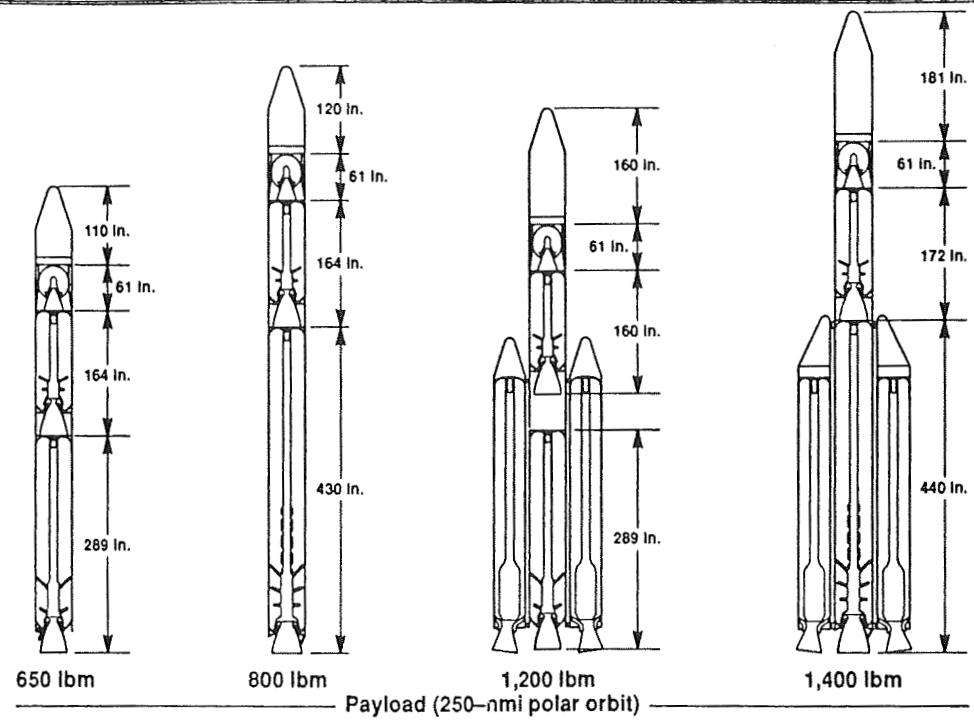
Independent Performance and Capability Distributions Combined Into One Failure Distribution: X=C–P



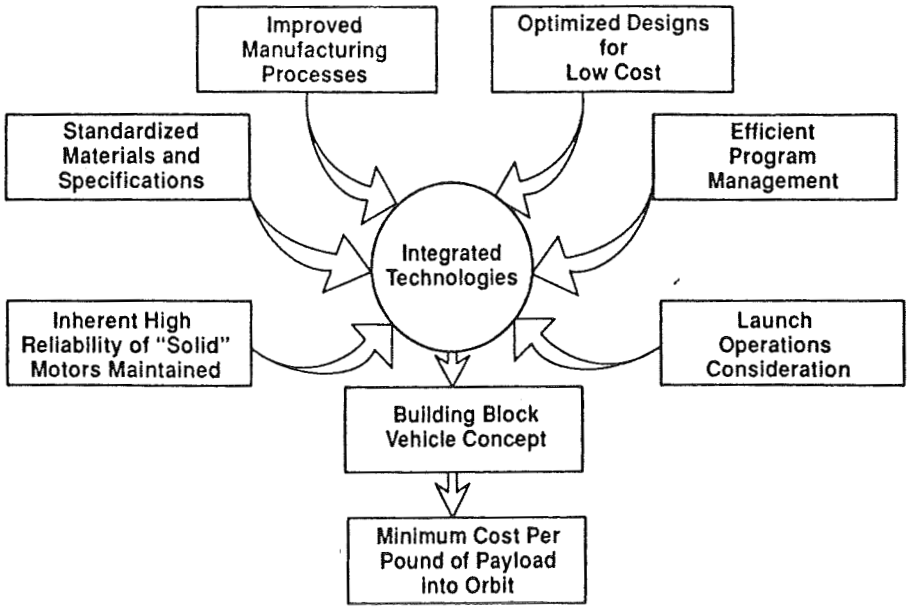
Small Launch Vehicle Concept Objectives

- Provide family of small launch vehicles to increase user flexibility in delivering a broad range of payloads (600 to 2,000 lb) into LEO
 - Remote sensing satellites
 - Communication and scientific research satellites
 - Recoverable capsules for industrial applications
- Retain high reliability of military systems
- Vehicle family based on basic motors (building blocks) derived from current strategic motor systems
- Minimize launch operations relating to vehicle
- Provide resiliency and responsiveness to launch on alert

Small Launch Vehicle Concept

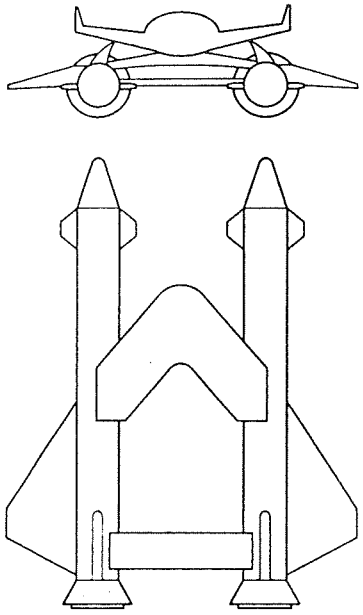


Small Launch Vehicle Enabling Technologies



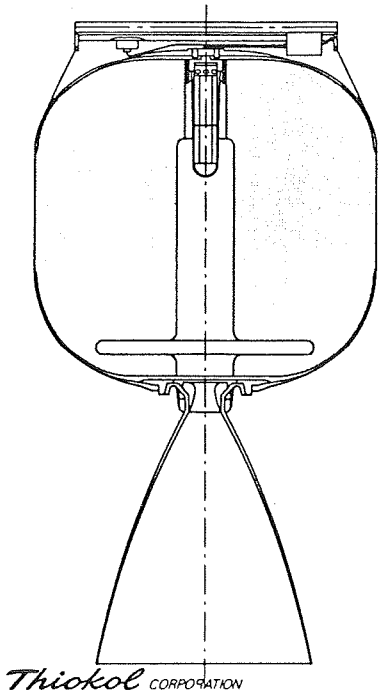
Thiokol CORPORATION

Reusable Flyback Booster System



- Concept objectives:
 - Solid rocket or hybrid propulsion
 - Booster transportation system for manned shuttle II and unmanned cargo carriers
 - Vertical launch, horizontal landing
 - Short turnaround cycle time
 - No preflight assembly required (load fuel and launch)
 - Lower recurring cost
- Enabling technologies:
 - Composite cases, struts, and wings
 - Cartridge–loaded propellant (SRM) or fuel (hybrid) grains
 - Integral removable aft dome/nozzle/skirt for quick fuel loading
 - Quick–change moldable nozzle insert or completely reusable (3–5 flights) advanced ceramic, passively cooled nozzle

High–Performance Solid Motors for Space



- Concept objectives
 - High–performance space propulsion system for:
 - Mars and lunar ascent propulsion
 - Orbit transfer propulsion
 - Long space storage capability
 - High I_{sp} performance
 - High mass fraction performance
- Enabling technologies
 - High–performance beryllium propellants
 - I_{sp} (theoretical) = 360–400 lbf–sec/lbm at 100:1
 - High propellant density (~0.05–0.06 lbm/in.^{3})
 - Braided carbon–carbon exit cone
 - 4D carbon–carbon throat
 - Consumable igniter
 - Laser–diode safe–and–arm device
 - Graphite composite case

Measured Comparison of Be and Al Propellants

Propellant	TP-H-3062	TP-H-1092
Metal fuel	Al	Be
Solids/metal (%)	86/16	86/12
Ballistics (BATES)		
Burn rate, 500 psi (in./sec)	0.246	0.260
Pressure exponent (n)	0.26	0.33
Theoretical I_{sp}, vac, $\varepsilon = 50$ (lbf-sec/lbm)	315.50	342.20
Measured I_{sp}, $\varepsilon = 50$ (lbf-sec/lbm)	293.00	312.50
Efficiency, η (%)	92.80	91.30

Conclusions

- Solids have multiple uses
 - Boosters
 - Small launch vehicles
 - Flybacks
 - Space transfer motors
- Keys to use
 - "Designed in" reliability
 - Low cost
 - Simplicity

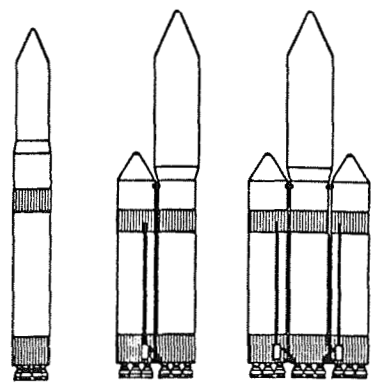
ADVANCED LAUNCH SYSTEM

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ADVANCED LAUNCH SYSTEM



SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM PENNSYLVANIA STATE UNIVERSITY

JAN C. MONK
GEORGE C. MARSHALL SPACE FLIGHT CENTER

June 27, 1990

U.S. SPACE TRANSPORTATION

COST
- $3,600 per POUND (AND UP)
- OVER $4B/YR

RELIABILITY
- OVER 5% FAILURES
- DOWNTIMES: UP TO 30+ MONTHS
- FAILURE COSTS ABOUT HALF LAUNCH COSTS

CAPABILITY
- SINGLE THREAD FOR CRITICAL PAYLOADS
- LITTLE OR NO MARGINS
- CONSTRAINTS INCREASE PAYLOAD COST
- BLOCKS FUTURE EXPANSION

ADVANCED LAUNCH SYSTEM

U.S. SPACE TRANSPORTATION (cont.)

INVESTMENT REQUIRED
- INFRASTRUCTURE OLD/MANPOWER INTENSIVE
- ELVs USE OLD DESIGNS/TECHNOLOGY

INFRASTRUCTURE NEEDS TO BE CHANGED
- OPERABILITY AND PERFORMANCE INADEQUATE FOR FUTURE NEEDS
- P3I HAS GOOD INTENTIONS AND POTENTAIL PAYOFFS, BUT THE INFRASTRUCTURE NEEDS TO BE CHANGED

"...BOOSTERS ARE REALLY TRUCKS...WE DON'T NEED A CADILLAC, MERCEDES, OR CORVETTE TO DELIVER OUR PACKAGES TO SPACE. WE NEED A VERY RELIABLE, MAINTAINABLE FLEET OF TRUCKS THAT CAN HAUL A VARIETY OF PACKAGES-QUICKLY AND CHEAPLY."

29 JULY 1988 E.A. ALDRIDGE, SEC AF

ADVANCED LAUNCH SYSTEM

WHAT IS ALS?

- A NEW WAY OF DOING BUSINESS
- A SYSTEM CONCEPT FOCUSED ON HIGH OPERABILITY AND LOW COST
- DEVELOPMENT, INTEGRATION AND TRANSFER OF NEW TECHNOLOGIES
- EFFECTIVE DEVELOPMENT AND USE OF INFORMATION SYSTEMS MANAGEMENT
- SUCCESSFUL APPLICATION OF TOTAL QUALITY MANAGEMENT (TQM)

Total Quality Management: THE OFFICIAL DOD DEFINITION

"Total Quality Management in the DOD is Strategy for continuously improving performance at every level, and in all areas of responsibility. It combines fundamental management techniques, existing improvement efforts, and specialized technical tools under a disciplined structure focused on continuously improving all processes. Improved performance is directed at satisfying such broad goals as cost, quality, schedule, and mission need and suitability. Increasing user satisfaction is the overriding objective." DOD 5000.51-G (DRAFT)

SIMPLY, TQM IS A MOVEMENT TO CURE THE TRADITIONAL MANAGEMENT PARADOX:



1. Improve quality
2. Costs decrease because of less rework, fewer mistakes, fewer delays, snags; better use of machine time and material
3. Productivity improves
4. Capture the market with better quality and lower price
5. Stay in business
6. Provide jobs and more jobs

-Dr. W. Edwards Deming

ADVANCED LAUNCH SYSTEM

DEFINITION OF QUALITY

- MEETING LAUNCH NEEDS AT THE LOWEST COST TO THE TAXPAYER
- ALS PROGRAM IS SYNONOMOUS WITH THE TQM GOALS
 - RELIABILITY
 - LOW COST
 - ROBUST

IF TQM DIDN'T EXIST -- WE'D INVENT IT

TQM IS ESSENTIAL TO THE SUCCESS OF ALS

ADVANCED LAUNCH SYSTEM

THE GOAL IS TO DEVELOP A ROBUST DESIGN

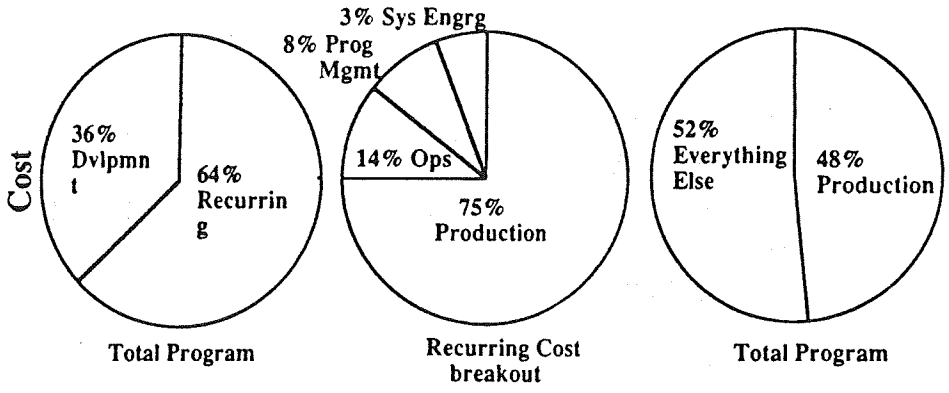
ALS QUALITY APPROACH

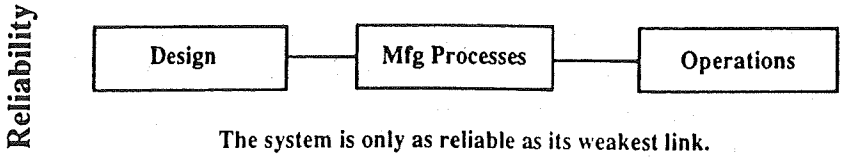
DESIGN IT RIGHT	*BUILD IT RIGHT*
CONCURRENT ENGINEERING VARIABILITY REDUCTION QUALITY FUNCTION DEPLOYMENT QUALITY ENGINEERING	IN-PROCESS MONITORING STATISTICAL PROCESS CONTROL CONTINOUS PROCESS IMPROVEMENT

ADVANCED LAUNCH SYSTEM

ALS Goals are Improved Reliability and Reduced Cost

Process Knowledge is a key to both areas.



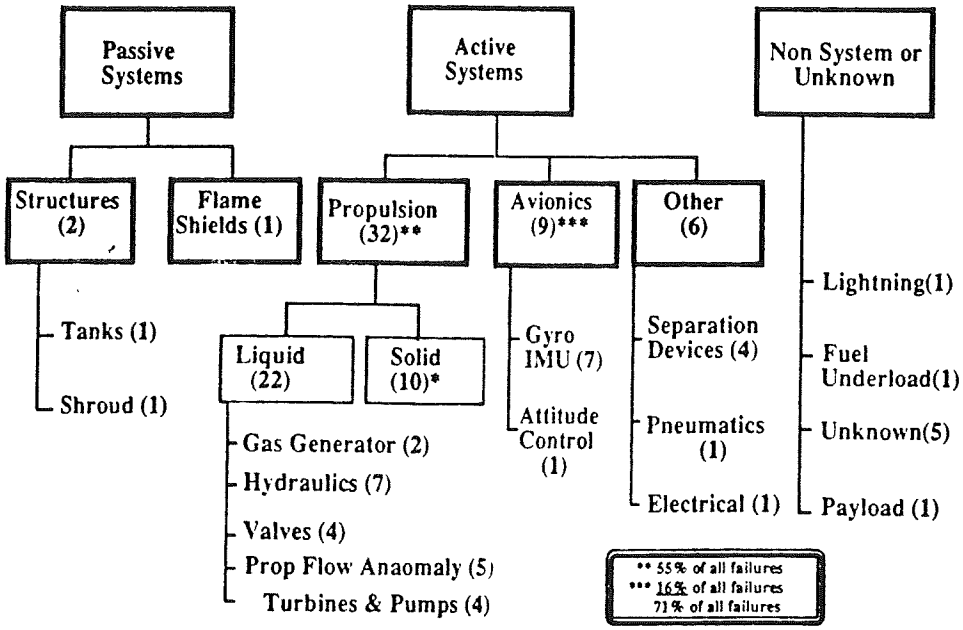


The system is only as reliable as its weakest link.

ADVANCED LAUNCH SYSTEM

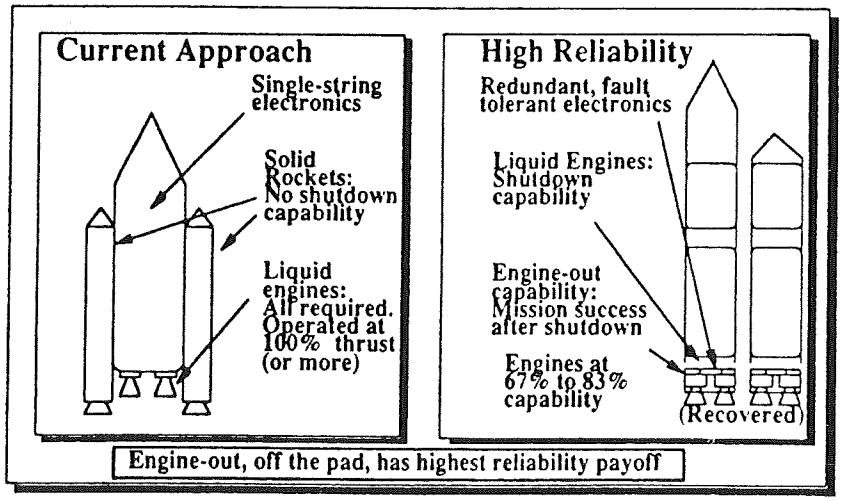
SUB SYSTEM SOURCES OF FAILURE

1966-1987

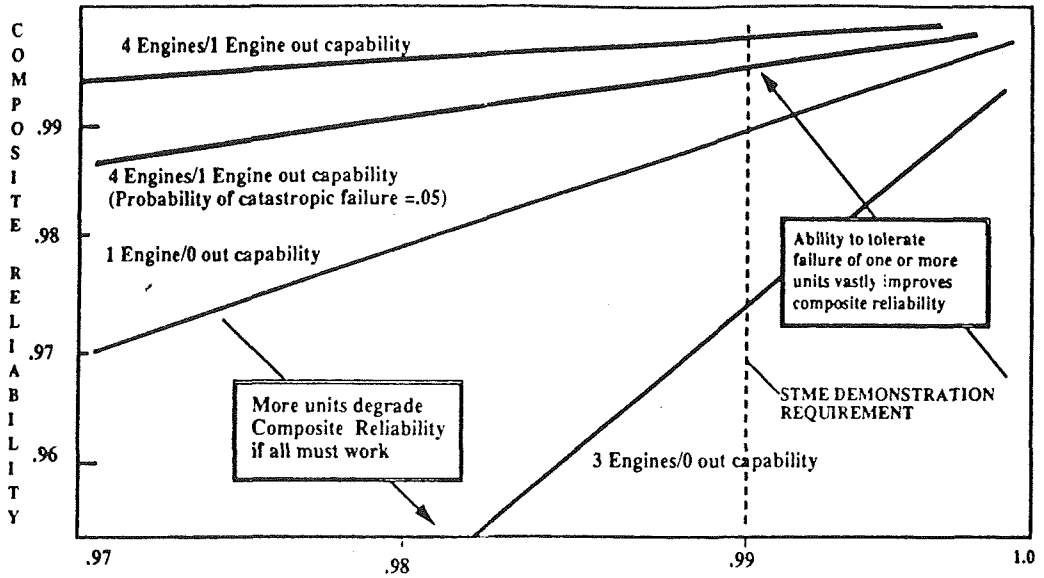


ADVANCED LAUNCH SYSTEM

DESIGN FOR RELIABILITY



Vehicle Engine Out Capability Provides A Significant Improvement In System Reliability



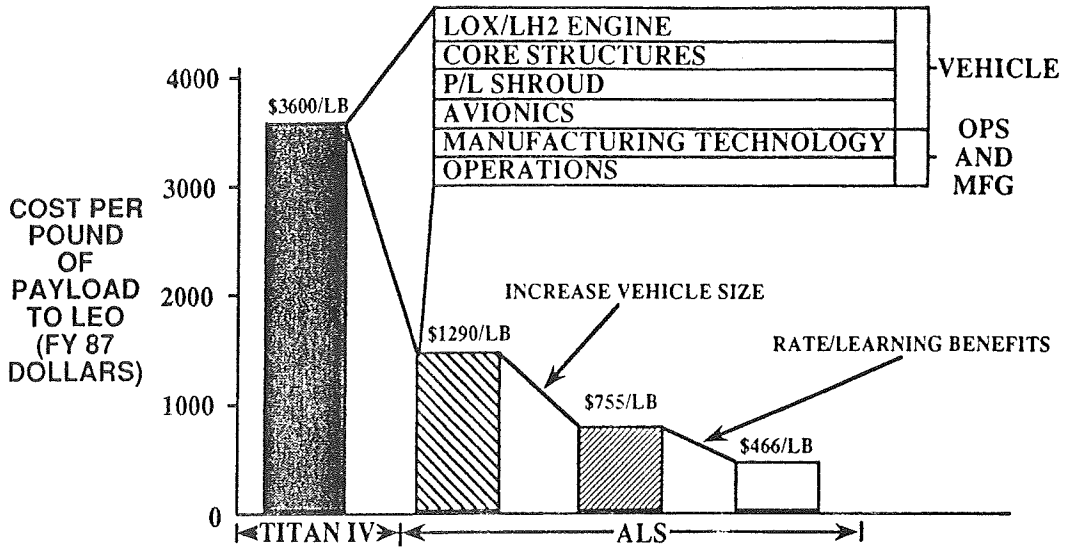
ADVANCED LAUNCH SYSTEM

FULL ENGINE-OUT CAPABILITY WITH HOLD DOWN FOR PRE-RELEASE ENGINE VERIFICATION SIGNIFICANTLY INCREASES FLIGHT RELIABILITY

- VEHICLE ENGINE OUT CAPABILITY REDUCES THE ENGINE RELIABILITY REQUIREMENT TO AN ACHIEVABLE VALUE
- ADDITIONAL RELIABILITY CAN BE ACHIEVED BY IMPLEMENTATION OF HOLD-DOWN
 - HISTORY INDICATES THAT 35-50 PERCENT OF ENGINE FAILURES OCCUR DURING START

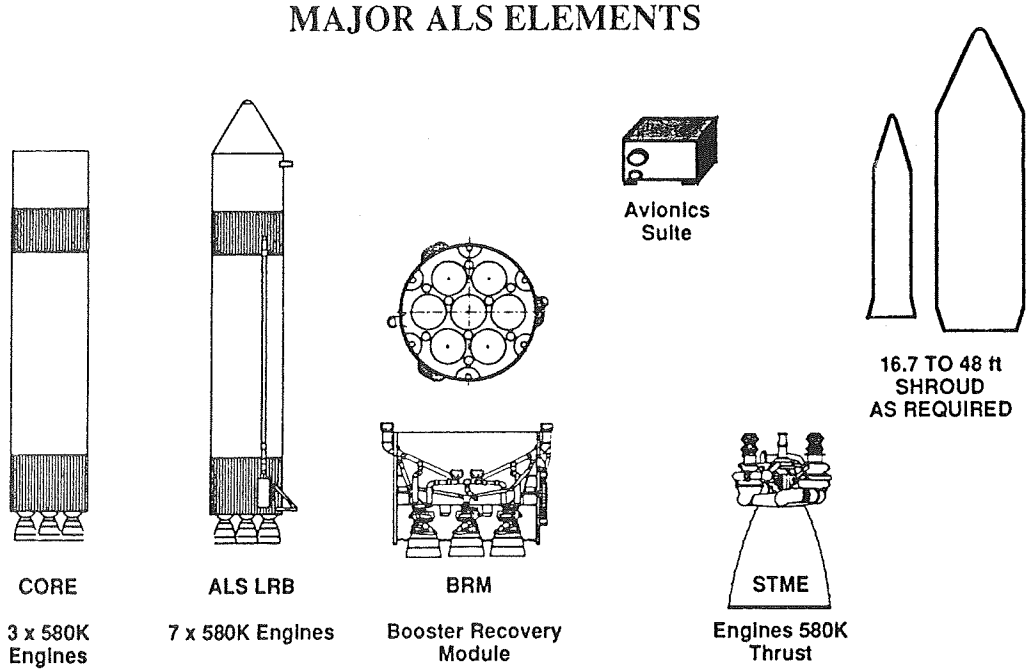
PATH TO REDUCED OPERATIONS COSTS

TECHNOLOGY BENEFITS



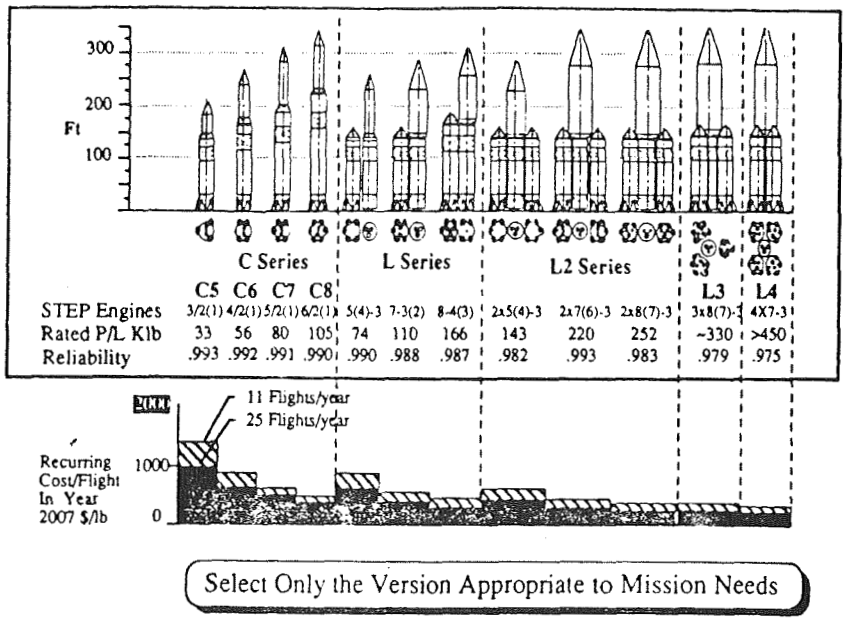
ADVANCED LAUNCH SYSTEM

MAJOR ALS ELEMENTS



ORIGINAL PAGE IS
OF POOR QUALITY

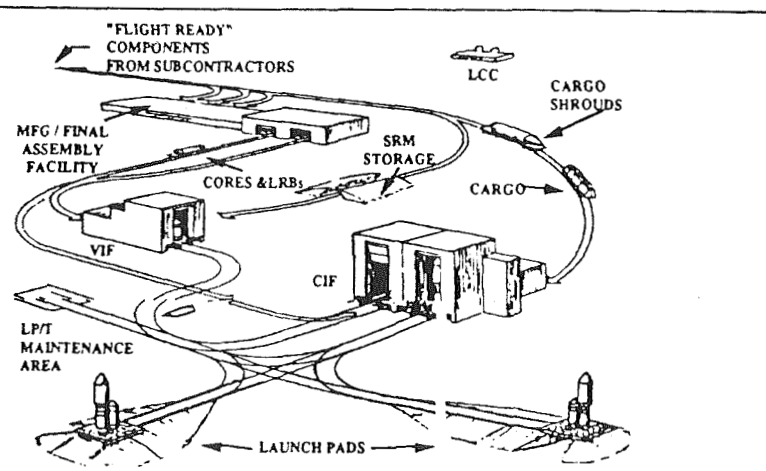
CANDIDATE ALS FAMILY MEMBERS



Select Only the Version Appropriate to Mission Needs

ADVANCED LAUNCH SYSTEM

KEY ALS FACILITIES



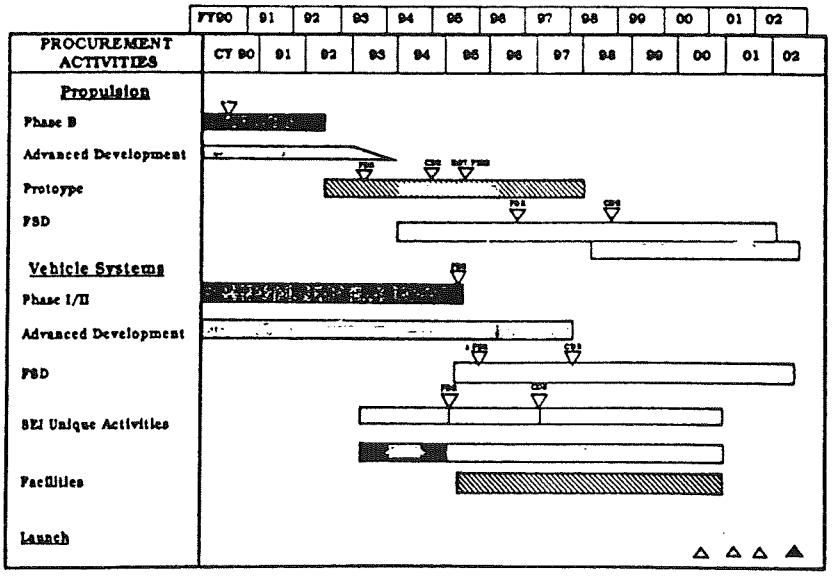
ALS Baseline Facilities Will Accommodate the Family

BALANCED ALS PROGRAM

- NEW PROPULSION SYSTEM (LONG LEAD)
 - BUILT WITH RELIABILITY, PERFORMANCE MARGIN, AND MAINTAINABILITY DESIGNED INTO THE SYSTEM
 - CONTINUE FIRST NEW INVESTMENT IN PROPULSION TECHNOLOGY IN MORE THAN A DECADE (AF AND NASA)
- NON-PROPULSION TECHNOLOGIES
 - SUPPORT FUTURE LAUNCH VEHICLE
 - SUPPORT FUTURE LAUNCH VEHICLE
 - COST / OPERABILITY
 - IMPROVE EXISTING LAUNCH VEHICLES
 - COST / OPERABILITY
 - PERFORMANCE

ADVANCED LAUNCH SYSTEM

ADVANCED LAUNCH DEVELOPMENT
PROGRAM SCHEDULE
(2000 ILC)



ALS CAN SATISFY THE NATION'S LAUNCH REQUIREMENTS BY PROVIDING A LOW COST, RELIABLE, ROBUST LAUNCH SYSTEM

"...BOOSTERS ARE REALLY TRUCKS...WE DON'T NEED A CADILLAC, MERCEDES, OR CORVETTE TO DELIVER OUR PACKAGES TO SPACE. WE NEED A VERY RELIABLE, MAINTAINABLE FLEET OF TRUCKS THAT CAN HAUL A VARIETY OF PACKAGES-QUICKLY AND CHEAPLY."

29 JULY 1988 E.A. ALDRIDGE, SEC AF

PRESENTATION 1.3.5

AIR FORCE SPACE SYSTEMS PROPULSION

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AFSC
AIR FORCE SYSTEMS COMMAND

SPACE SYSTEMS PROPULSION TECHNOLOGY

by: Dale Hite

SPACE SYSTEMS PROPULSION TECHNOLOGY VISION

AFSC
AIR FORCE SYSTEMS COMMAND

NEEDS

- FULL OPERATIONAL USE OF SPACE DICTATES MULTI-PURPOSE VEHICLES THAT ASSUME:
 - MILITARY RESPONSIVENESS
 - RELIABILITY/MAINTAINABILITY
 - SUPPORTABLE LOGISTICS
 - TECHNOLOGICAL INNOVATION

ELEMENTS

- SSTO
- ORBIT TRANSFER AND MANEUVERING
- REVOLUTIONARY CAPABILITY

ORBIT TRANSFER AND MANEUVERING PROPULSION

AFSC
AIR FORCE SYSTEMS COMMAND

CHEMICAL PROPULSION

- MODULAR STORABLE PROPULSION - 8,000 TO 10,000 LBM TO GEO; IMBEDDED MANEUVERING PROPULSION PACKAGING FLEXIBILITY AND UNLIMITED RESTART - BY FY 92
- ROCKET ENGINE MATERIALS - 50% WEIGHT REDUCTION WITH ELIMINATION OF CORROSION AND WEAR LIMITS ON RELIABILITY AND COST - BY FY 93
- CRYOGENIC ENGINE TECHNOLOGY - 13,000 TO 15,000 LBM TO GEO - BY FY 93

MODULAR/STORABLE ORBIT TRANSFER/MANEUVERING PROPULSION

AFSC
AIR FORCE SYSTEMS COMMAND

FEATURES

- High Performance Storable Engines (XLR-132)
- Modular, Compact Propellant Feed Systems
- Payload Increase to GEO
 - 45% With Titan IV/IUS
 - More than 75% with Shuttle
- 40% More Maneuvering Capability for Survivability
- 50% Volume Savings

TRANSITION TARGETS

- 10K Class ELV Upper Stages
- GPS, DSCS, DSP (Block Changes)

CRYOGENIC OTV PROPULSION

AFSC
AIR FORCE SYSTEMS COMMAND

FEATURES

- 29% More Payload Volume than Centaur G' (40 FT Payload)
- Capable of 30 Day Hold in LEO
- Enable 40% Payload Increase to GEO

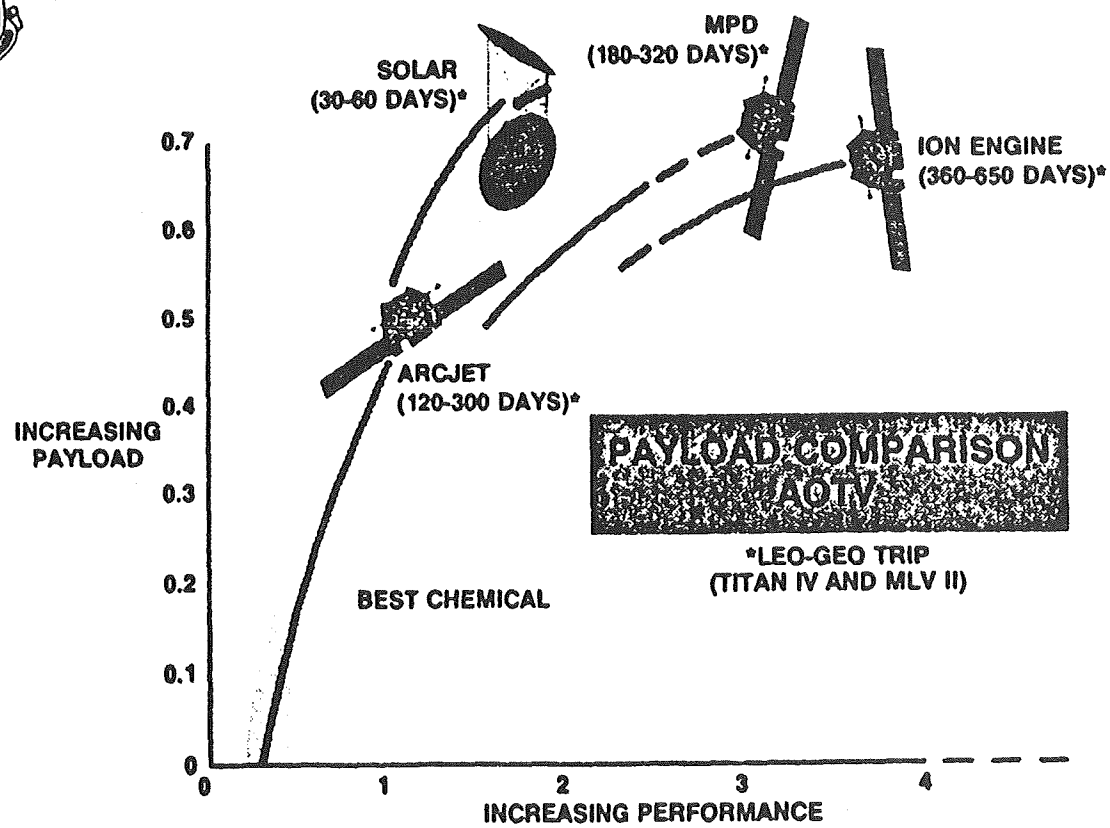
TRANSITION TARGETS

- 13K+ Class ELV Upper Stages

ADVANCED SPACE PROPULSION

- ELECTRIC PROPULSION - ARCJET 60 TO 100% INCREASE IN PAYLOAD; 12,000 LBS OFF TITAN AND 5,000 LBS OFF ATLAS - BY FY 2000
- ELECTRIC PROPULSION - MAGNETOPLASMADYNAMIC THRUSTER - BY FY 2025
- SOLAR PROPULSION - 26,000 TO 45,000 LBS FROM LEO TO GEO, SHUTTLE LAUNCH; ELIMINATES SOLAR ARRAY PROBLEMS - BY FY 2010

PAYLOAD AND TRIP TIME



ELECTRIC VERSUS CHEMICAL PROPULSION

AFSC
AIR FORCE SYSTEMS COMMAND

ELECTRIC (30KWE ARCJET)

- PROPELLANT CONSUMPTION 1/2 TO 1/3 THAT OF CHEMICAL
- LARGE POWER SOURCE REQUIREMENT
- FIVE MONTH TRANSFER TIMES

CHEMICAL

- HIGH PROPELLANT CONSUMPTION
- STORED CHEMICAL ENERGY SOURCE
- SHORT TRANSFER TIMES

HIGH ENERGY DENSITY MATTER

AFSC
AIR FORCE SYSTEMS COMMAND

IDENTIFICATION AND SYNTHESIS

- IN-HOUSE CENTER OF EXCELLENCE - CONTINUING EFFORT

PROPELLANT DEVELOPMENT - FY 2005

- DOUBLE PAYLOADS
- REDUCE ROCKET SIZE BY 50%
- SINGLE STAGE TO ORBIT

OTHER TECHNOLOGY BASE ACTIVITIES

- DOD CRITICAL TECHNOLOGIES LIST
- AFOSR/NC
- NASA LEWIS
- WRDC/POOC

NUCLEAR UPPER STAGE OFF ATLAS II

- **TECHNOLOGY DEMONSTRATED**
 - ACHIEVED ASSEMBLY OF POWER SYSTEM AT SANDIA
 - MEASURED POWER SYSTEM HEATING RATES AT BROOKHAVEN
 - DEMONSTRATED STRUCTURAL INTEGRITY OF POWER SYSTEM AT LOW POWER AND NOMINAL TEMPERATURES
- **PAYOFF**
 - 400% INCREASE OVER CENTAUR G OFF ATLAS II

SSTO PROPULSION TECHNOLOGY

- **ALTITUDE COMPENSATING NOZZLE - 15 SECOND ISP GAIN AND 35% PAYLOAD GAIN - BY FY 91**
- **ADVANCED MATERIALS FOR NASP PROPULSION SYSTEM - $30M COST SAVINGS, 2,000 LB WEIGHT SAVINGS - ELIMINATE POTENTIAL ENVIRONMENTAL CONCERN - BY FY 92**
- **LONG LIFE COMBUSTION CHAMBER - 300 CYCLES - BY FY 92**
- **INJECTOR SPRAY CHARACTERIZATION:**
 - MANIFOLD AND ORIFICE HYDRAULICS - BY FY 91
 - ATOMIZATION AND MIXING - BY FY 92
 - SECONDARY DROPLET BREAKUP - BY FY 93
 - SUPERCRITICAL VAPORIZATION - BY FY 94

ADVANCED LAUNCH DEVELOPMENT PROGRAM TECHNOLOGIES AT AL

AFSC
AIR FORCE SYSTEMS COMMAND

FEATURES

- ALTERNATE PYROTECHNICS
- LOW COST EXPENDABLE CRYOGENIC TANKS
- CLEAN PROPELLANT DEVELOPMENT
- SOLID PROPULSION INTEGRATION AND VERIFICATION
- NON-DESTRUCTIVE EVALUATION OF SOLID ROCKET BOOSTERS
- LOW COST EXPENDABLE LOX/H2 ENGINE DEVELOPMENT
- ROCKET ENGINE CONDITION MONITORING

AL SUPPORT TO NASP

AFSC
AIR FORCE SYSTEMS COMMAND

TECHNICAL MANAGEMENT

- ROCKET PROPULSION TECHNOLOGY
- ONE CO-LOCATED AL ENGINEER WITH JOINT PROGRAM OFFICE

TECHNICAL SUPPORT

- AIR BREATHING ENGINES

PLANNING SUPPORT

- FEED SYSTEMS AND TURBOPUMPS

POTENTIAL SUPPORT

- TESTING FACILITIES AND FLIGHT TESTING AT AFFTC

PRESENTATION 1.3.6

UNMANNED LAUNCH VEHICLES / UPPER STAGES

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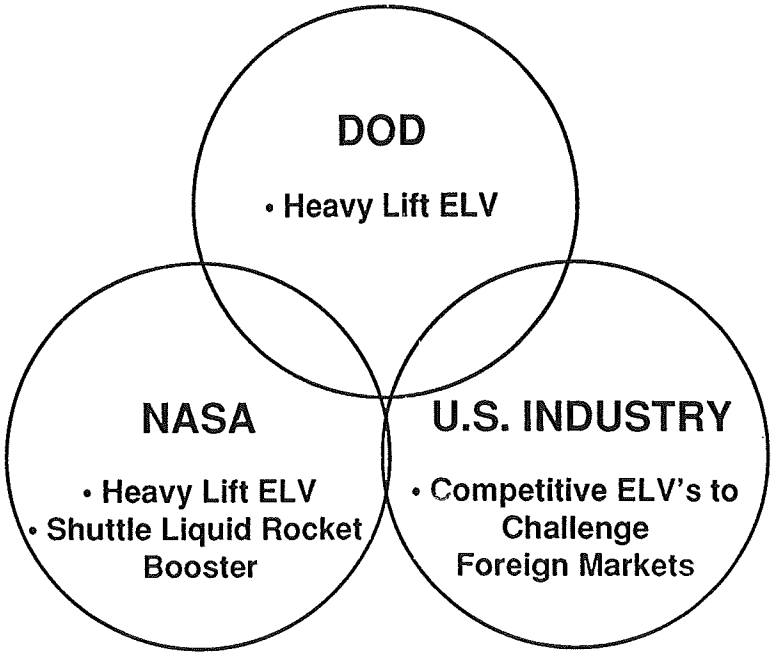
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NEXT GENERATION

UNMANNED LAUNCH VEHICLES AND UPPER STAGES

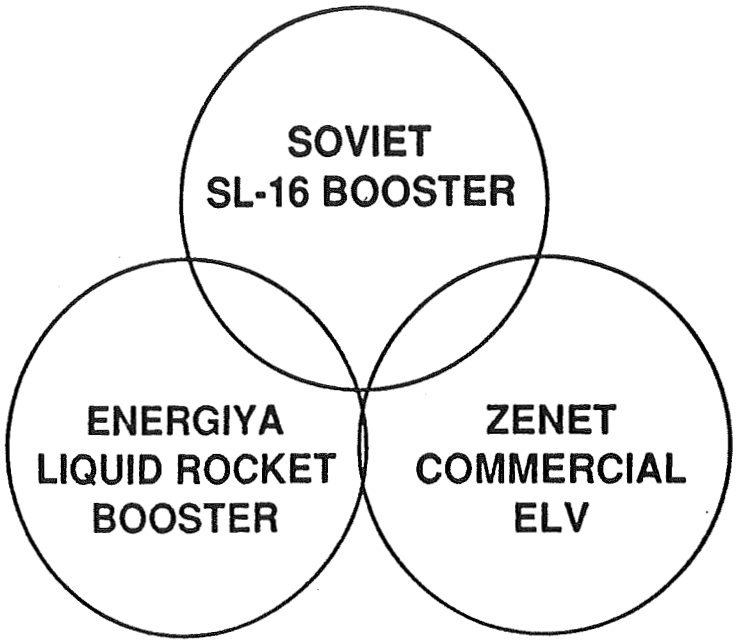
CHARLES R. GUNN
NASA
OFFICE OF SPACE FLIGHT
JUNE 27, 1990

THE NEEDS



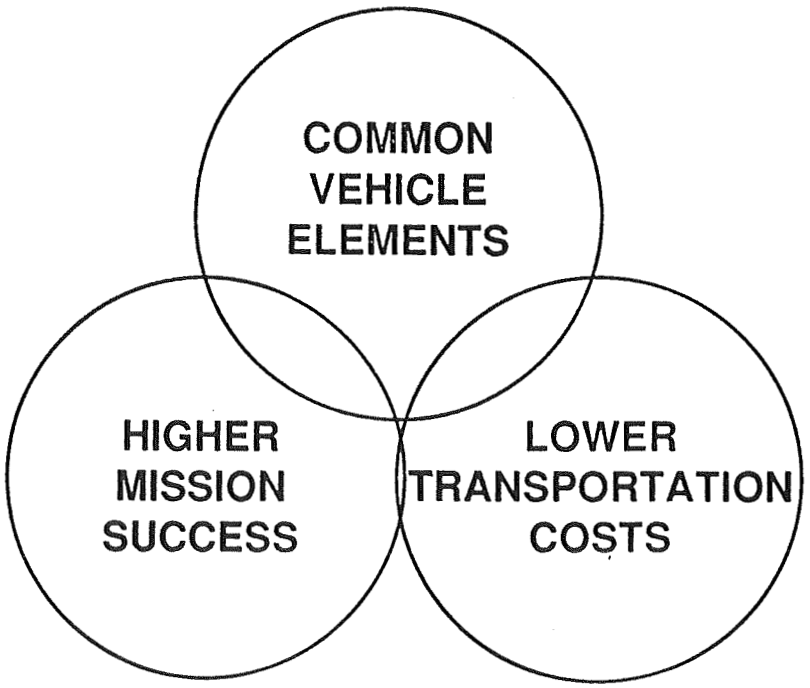
A NATIONAL CONSORTIUM TO DEVELOP AND PRODUCE COMMON VEHICLE ELEMENTS

THE MODEL



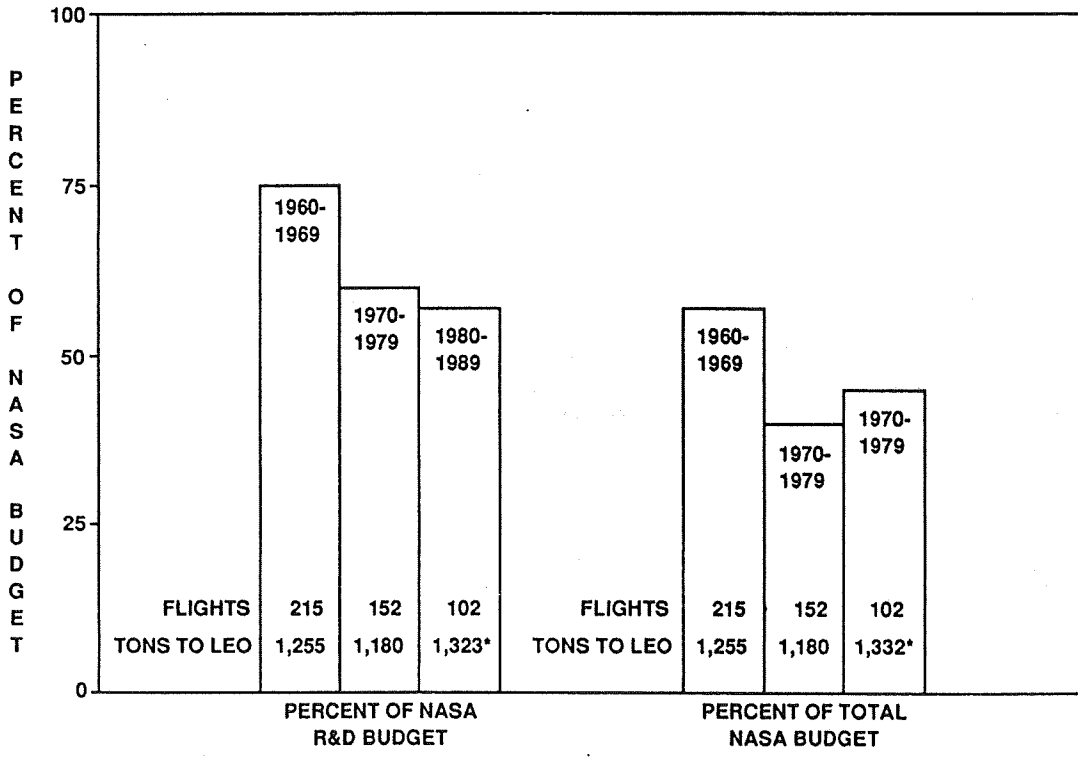
SOVIET ZENET FULFILLS
MILITARY, SPACE AND COMMERCIAL NEEDS

THE FOCUS



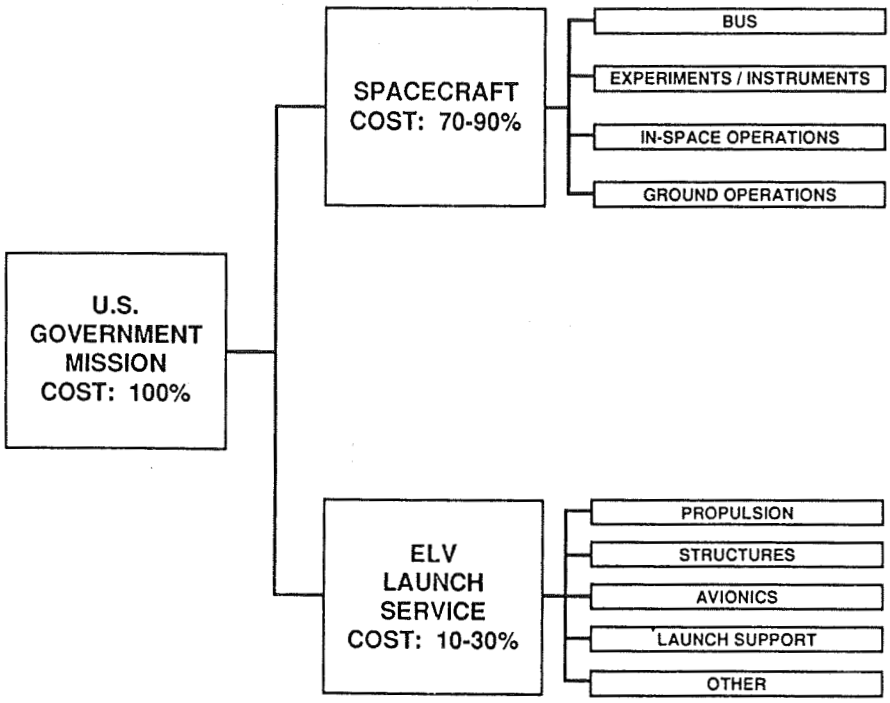
LOWER TRANSPORTATION COST

NASA SPACE TRANSPORTATION RESOURCES



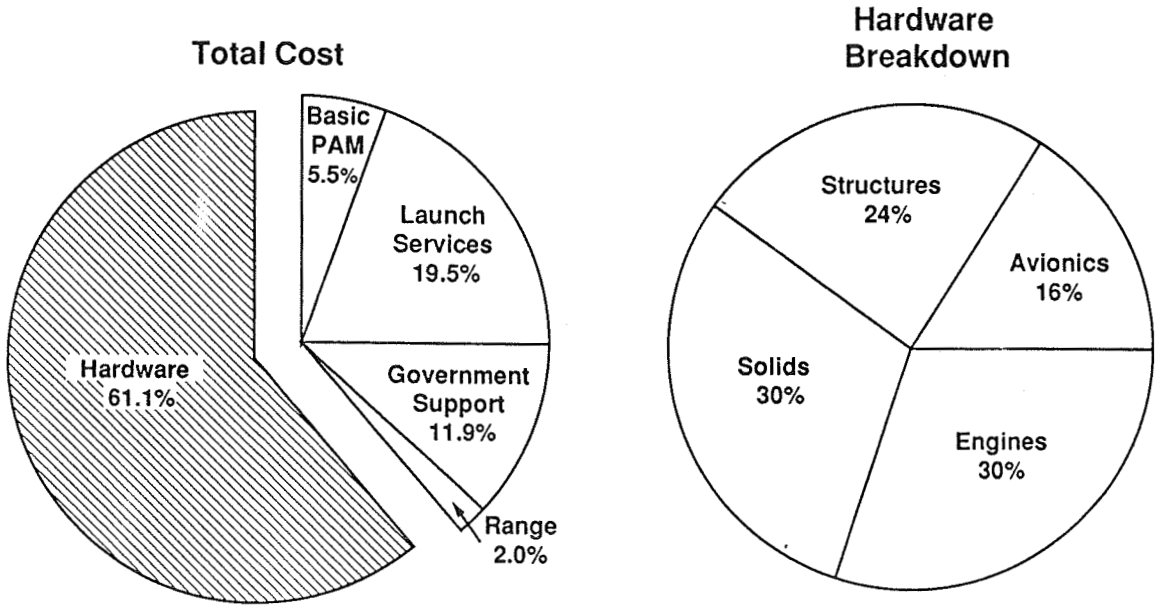
* W/O SPACE SHUTTLE ORBITER (3,763 TONS INCLUDING ORBITER)

PERSPECTIVE ON MISSION COST



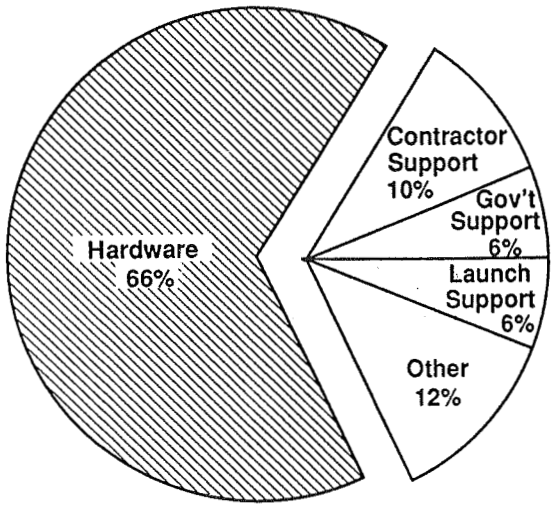
NOTE: U.S. GOVERNMENT MISSIONS ON MEDIUM & LARGE PERFORMANCE CLASS ELV's (e.g., DELTA II AND LARGER)

DELTA 7925 - RECURRING COST

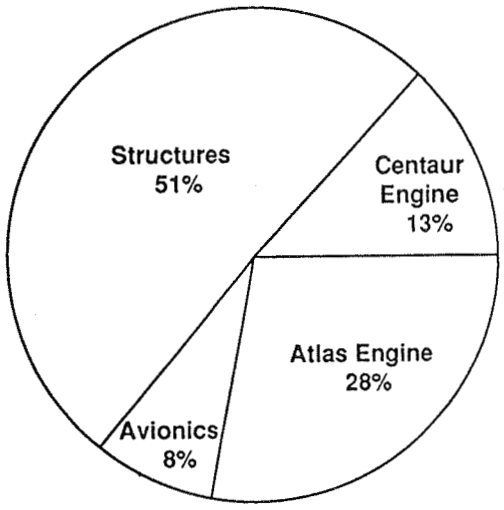


ATLAS / CENTAUR - RECURRING COST
(4 FLIGHTS / YEAR)

Total Cost

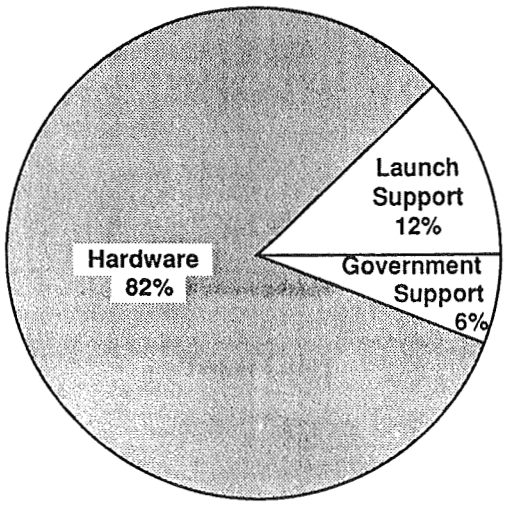


Hardware
Breakdown

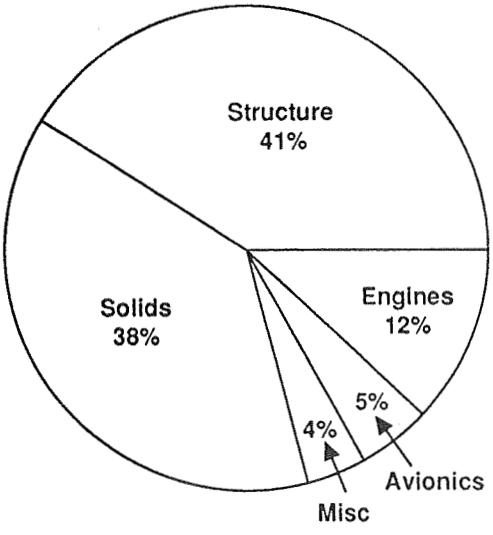


TITAN III - RECURRING COST

Total Cost

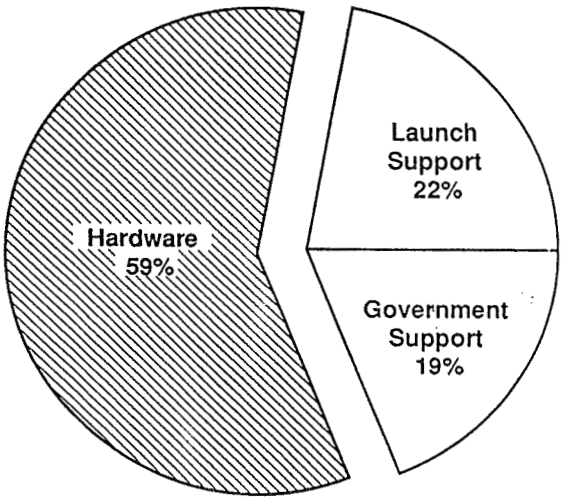


Hardware
Breakdown

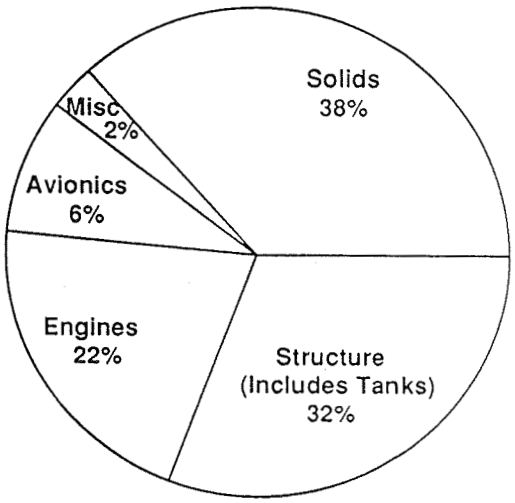


TITAN IV - RECURRING COST
(4 FLIGHTS / YEAR)

Total Cost



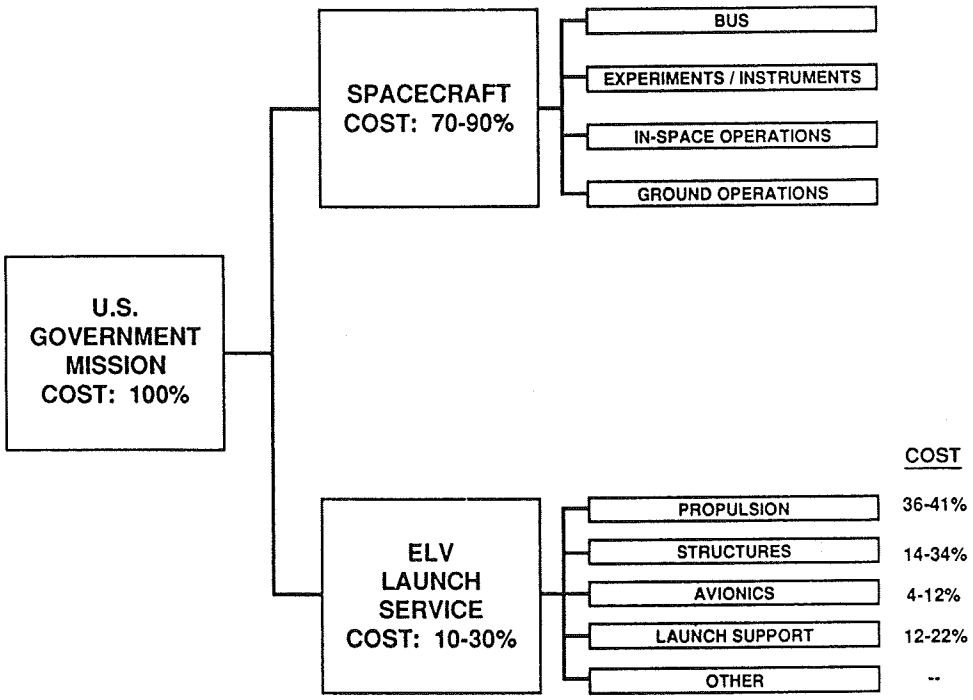
Hardware
Breakdown



ENGINE COSTS

ENGINE	THRUST K LBS	PROPELLANTS	COST, FY 1990$
SSME (STS)	470	H_2 / O_2	$44M - Each (Quantity of 4)
MB-3 SET (ATLAS II)	423 / 85	RP-1 / O_2	$13 - 14M - Set (Quantity of 18)
RS-27 (DELTA II)	237	RP-1 / O_2	$8 - 9M - Each (Quantity of 20)
RL-10 SET (CENTAUR)	33	H_2 / O_2	$4½ - 5M - Set (Quantity of 20)
VIKING-VI (ARIANE IV)	150	A-50 / N_2O_4	$4 - 5M - Each (Quantity of ?)
VIKING HM-60 (ARIANE V)	250	H_2 / O_2	--

PERSPECTIVE ON MISSION COST

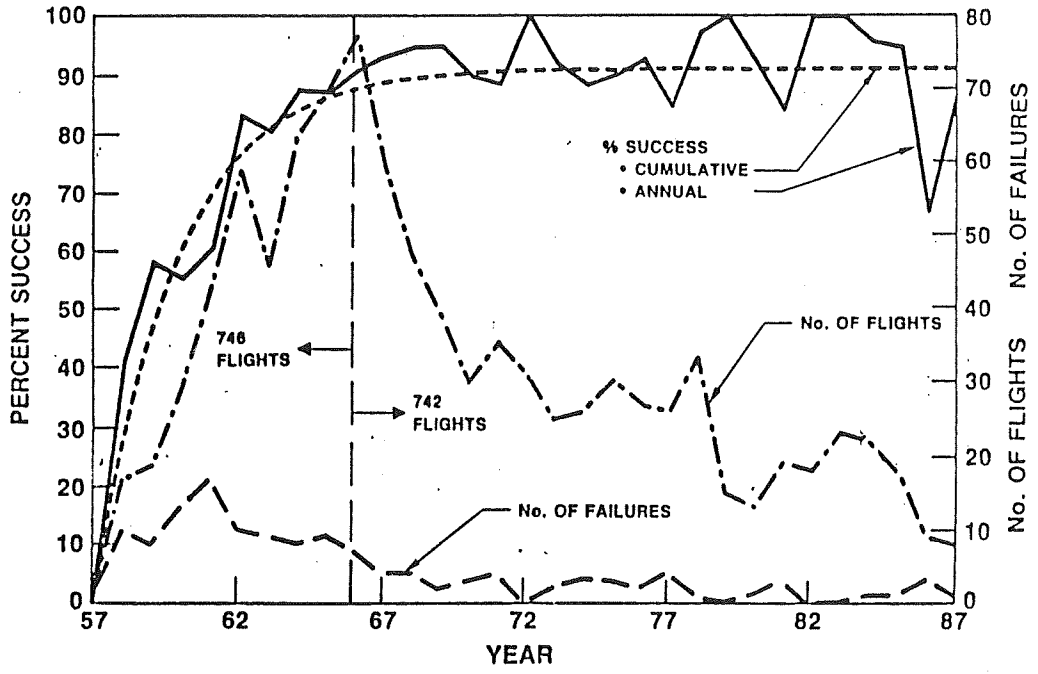


NOTE: U.S. GOVERNMENT MISSIONS ON MEDIUM & LARGE PERFORMANCE CLASS ELV's (e.g., DELTA II AND LARGER)

HIGHER MISSION SUCCESS

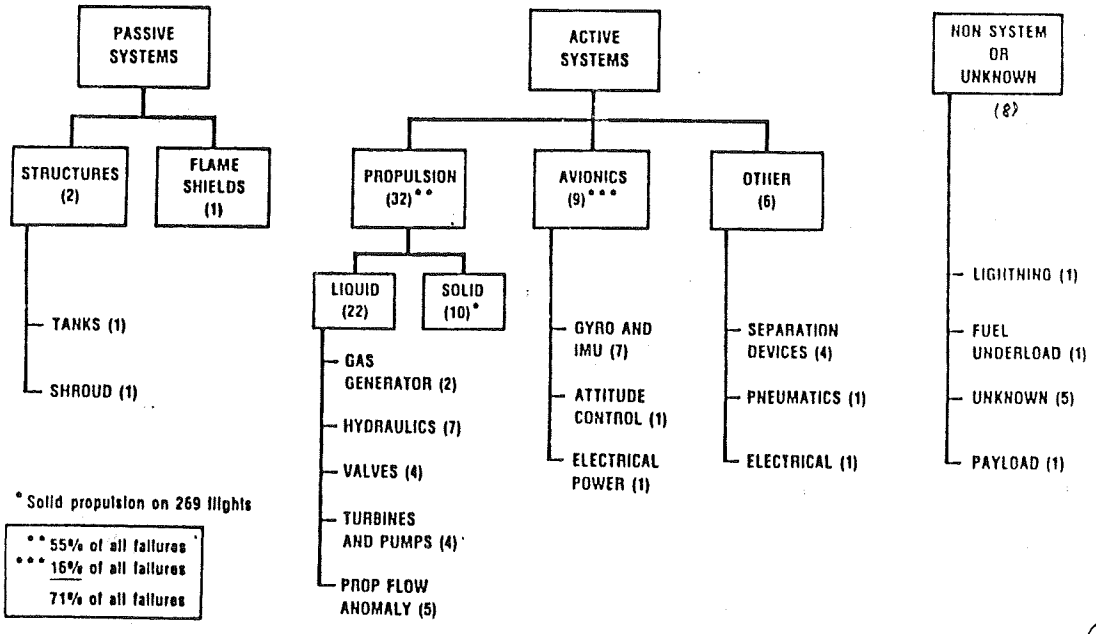
U.S. Launches, 1957–1987

VANGUARD, JUPITER, THOR/DELTA, JUNO, ATLAS, SCOUT, REDSTONE, SATURN, TITAN, STS

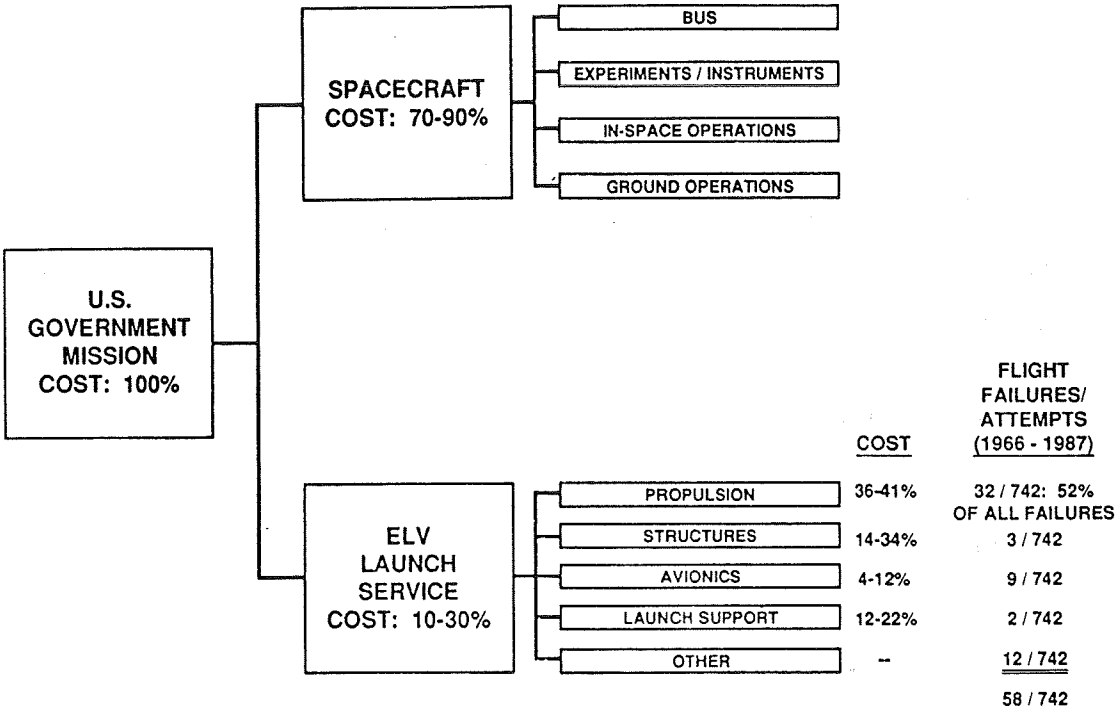


Subsystem Sources of Failure

1966–1987
742 TOTAL FLIGHTS
(1966–1987) — ATLAS, THOR/DELTA, TITAN, SCOUT, STS
58 FAILURES



PERSPECTIVE ON MISSION COST AND FAILURES



NOTE: U.S. GOVERNMENT MISSIONS ON MEDIUM & LARGE PERFORMANCE CLASS ELV's (e.g., DELTA II AND LARGER)

SUMMARY OF FLIGHT EXPERIENCE

- **PROPULSION SYSTEM COSTS ARE LARGEST FRACTION OF ELV (35%)**
- **PROPULSION SYSTEMS HAVE HIGHEST FAILURE RATE**
 - 52% OF ALL FAILURES
 - >50% OF FAILURES ATTRIBUTED TO POOR WORKMANSHIP OR HUMAN ERROR
- **LIQUID ENGINE FAILURES**
 - 1/3 IN ENGINE - (NO CRYO ENGINE FAILURE)
 - 2/3 IN ASSOCIATED SYSTEMS (FEED LINES, VALVES, PRESSURIZATION SYSTEM, ACTUATORS, HYDRAULIC PUMP, ETC.)
 - 75% OF ALL ENGINE FAILURES OCCUR AT STARTUP
- **PROPULSION SYSTEM BENIGN TO CATASTROPHIC FAILURE RATIO 10:1**
 - ENGINE OUT CAPABILITY WOULD HAVE INCREASED MISSION SUCCESS
 - HIGH RELIABILITY ENGINE INSTRUMENTATION ESSENTIAL

RECOMMENDATIONS FOR NEXT GENERATION SPACE TRANSPORTATION

- **ESTABLISH A NATIONAL CONSORTIUM:**
 - AGGREGATE NASA / DOD / ELV COMMERCIAL INDUSTRY REQUIREMENTS
 - AGREE ON COMMON PROPULSION ELEMENTS
 - **ENGINE**
 - **PROPELLANT TANK MODULES**
 - **PRESSURIZATION SYSTEM**
 - **THRUST VECTOR CONTROL SYSTEM**
 - **SYSTEM MANAGEMENT SCHEME**
 - AGREE ON SHARING OF:
 - **MANAGEMENT**
 - **NON-RECURRING COSTS**
 - **PRIORITY OF PRODUCTION / LAUNCH ASSETS**
 - **FLIGHT FAILURES CORRECTIVE ACTIONS**

RECOMMENDATION FOR NEXT GENERATION SPACE TRANSPORTATION (CONTINUED)

- **FOCUS MORE DESIGN ENGINEERING ON ENGINE SUPPORT SYSTEMS**
 - 2/3 OF PROPULSION SYSTEMS FLIGHT FAILURES
- **REASSESS PROGRAM MANAGEMENT OF NEXT ENGINE DEVELOPMENT - FRESH PERSPECTIVE ON:**
 - MISSION SUCCESS vs HIGHEST PERFORMANCE
 - PRODUCIBILITY vs LOWEST WEIGHT AND SMALLEST ENVELOPE
 - DURABILITY vs FREQUENT FIELD CHANGE-OUT
 - REUSEABLE vs EXPENDABLE

CHALLENGE THE INDUSTRY
(SPACE AND AIRCRAFT ENGINE MANUFACTURERS)

- **$100K AND 12 MONTHS TO DESIGN AND BUILD A 250 KLB THRUST H_2 / O_2 ENGINE**
- **U.S. GOVERNMENT TO CONDUCT TEST FIRE DEMONSTRATION**

LOW COST ENGINE DEMONSTRATION

- **TO BUILD A 250K LB THRUST H_2 / O_2 ENGINE FOR $100K IN 12 MONTHS MUST:**
 - "CHARGE" THE TEAM - THEN HANDS-OFF AND LET TEAM WORK. RECOGNIZE ACCOMPLISHMENTS
 - FORM SMALL "CAN-DO" TEAM AT A SINGLE LOCATION - THE RIGHT PEOPLE
 - BREAKOUT OF CURRENT HIGH TECH, HIGH COST, COMPLEX AEROSPACE CULTURE
 - KEEP EFFORT SMALL AND MANAGEMENT SIMPLE - AVOID TIME CONSUMING, COSTLY BUREAUCRACY AND REPORTING

LOW COST ENGINE DEMONSTRATION

- **DEMONSTRATE ENGINE WORTHINESS**
 - RIGHT PEOPLE AND WORK ENVIRONMENT AS ABOVE
 - INSTRUMENT ENGINE
 - MAXIMUM STARTS AND RUN TIME
 - RUN TO FAILURE / IMPENDING FAILURE
- **RESULTS**
 - CASE I - ENGINE SUCCESSFULLY STARTS AND ACCUMULATES LONG RUN TIME WITHOUT MAJOR PROBLEMS

 RESULT - **LOW COST ENGINE METHODS, TECHNIQUES, HARDWARE DEMONSTRATED**
 - CASE II - ENGINE FAILS EARLY

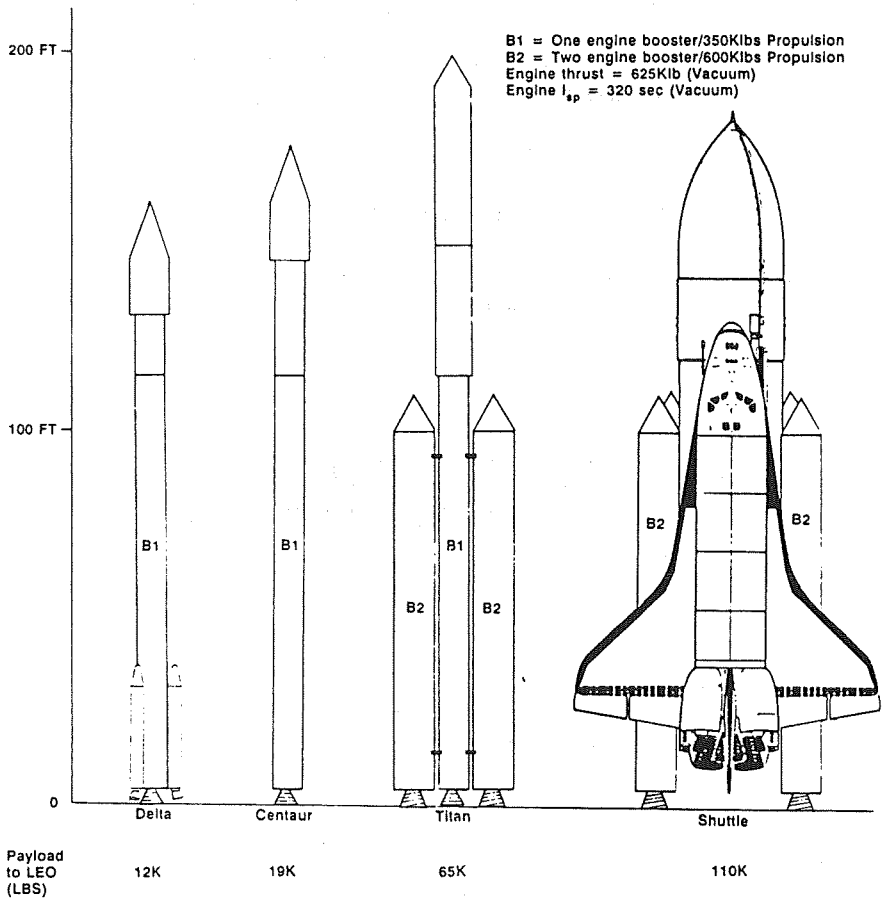
 ACTION - **DETERMINE CAUSE AND CORRECTIVE ACTION**

 RESULT - **HARD FACTS ON PITFALLS TO AVOID IN LOW COST ENGINE - HOW TO DO IT RIGHT**

NEXT GENERATION COMMERCIAL ELV NEEDS ESTIMATE
(PROPULSION ONLY)

- **BOOSTER**
 - **CAPABILITY TO LEO 50 - 70K LBS**
 - **500 - 600K LBS THRUST LEVEL CORE ENGINES**
 - **ENGINE SYSTEM - OUT CAPABILITY**
 - **CLEAN PROPELLANTS - H_2 / O_2 OR HYDRO CARBONS / O_2**
 - **STAND ALONE STRUCTURE**
 - **14 - 18 FEET DIAMETER**
 - **90 - 110 FEET LONG**
 - **MODULAR STRAP-ON LIQUID / SOLID ROCKET MOTORS CAPABILITY**
 - **RECOVERABLE OPTION**
 - **LOW COST - MAX $20M IN FY 1990 $ FOR TOTAL BOOSTER**
 - WITH LIQUID / SOLID ROCKET MOTORS
 - BLOCK BUY OF 20
- **SECOND STAGE**
 - **CAPABILITY TO GTO 15 - 20K LBS**
 - **35 - 45K LBS THRUST LEVEL CORE ENGINES**
 - **ENGINE SYSTEM-OUT CAPABILITY**
 - **H_2O_2 PROPELLANTS**
 - **STAND ALONE STRUCTURE**
 - **14 - 18 FEET DIAMETER**
 - **LOW COST - MAX. $25M IN FY 1990 $ FOR TOTAL STAGE**

Single LOX/RP1 Engine
One Engine & Two Engine Booster Stages

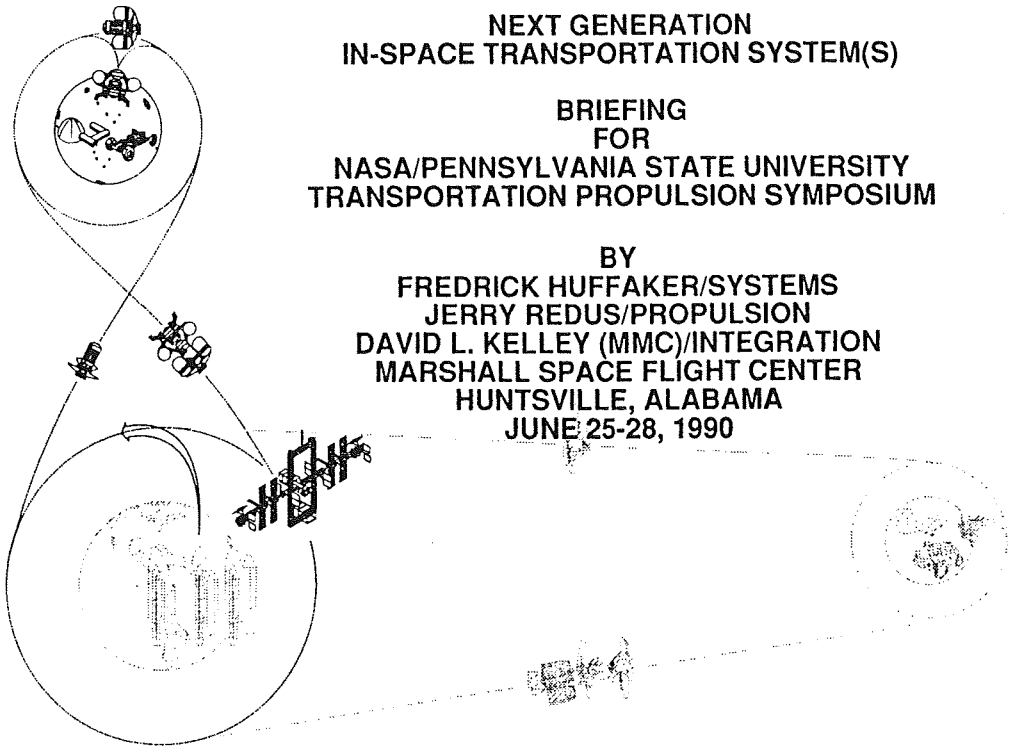


SPACE TRANSFER VEHICLES

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NEXT GENERATION IN-SPACE TRANSPORTATION SYSTEM(S)

BRIEFING
FOR
NASA/PENNSYLVANIA STATE UNIVERSITY
TRANSPORTATION PROPULSION SYMPOSIUM

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JUNE 25-28, 1990

NEXT GENERATION
IN-SPACE TRANSPORTATION SYSTEM

Abstract

The development of the next generation In-Space Transportation System presents a unique challenge to the design of a propulsion system for the Space Exploration Initiative (SEI). Never before have the requirements for long-life, multiple mission use, space basing, high reliability, man-rating and minimum maintenance come together with performance in one system that must protect the lives of our space travelers, support the mission logistics needs and do so at an acceptable cost. The challenge before us is to quantify the bounds of these requirements. The issue is one of degree. How long is an acceptable life in space? When does reuse pay off? To what degree is space basing practical; full, partial or expended? These are issues that determine the reusable bounds of a design and include dependability, contingency capabilities, resiliency and minimum dependence on a maintenance node in preparation for and during a mission. Missions to planet earth, other non-NASA missions and planetary missions will provide important but less demanding requirements for the transportation systems of the future.

The missions proposed for the Space Exploration Initiative will require a family of transportation vehicles to meet the requirements for establishing a permanent human presence on the moon and eventually on Mars. Specialized vehicles will be needed to accomplish different phases of each mission. These large scale missions will require assembly in space and will provide the greatest usage of the planned integrated transportation system.
This paper looks at the current approach to defining the In-Space Transportation System for the SEI moon missions with later Mars mission applications. It reviews several system development options, propulsion concepts, current / proposed activities and outlines key propulsion design criteria, issues and technology challenges for the next generation In-Space Transportation System(s).

NEXT GENERATION IN-SPACE TRANSPORTATION SYSTEM

AGENDA

- INTRODUCTION
- PROGRAM SCHEDULE (PLANNING)
- IN-SPACE TRANSPORTATION SYSTEM INTERFACES
- LUNAR TRANSPORTATION REQUIREMENTS (OPTION 5)
- PROGRAM DEVELOPMENT OPTIONS / CURRENT CONCEPT STATUS
- PROPULSION CONCEPT APPROACH
- ENGINE DEVELOPMENT CRITERIA
- TECHNOLOGY ISSUES
- PROPOSED ACTIVITIES
- CHALLENGE / SUMMARY

SPACE EXPLORATION INITIATIVE PROGRAM SCHEDULE (Preliminary)

The Space Exploration Initiative proposed by President Bush will expand human presence and activity into the solar system including the moon and Mars. This means permanent human presence in space. Several architectures are currently under evaluation by NASA to determine how to best accomplish this objective. A reference architecture, Option 5, was used in our ' 90 Day In-House Study ' activities in 1989. The Option 5 schedule is Figure 1 and shows the milestones for the Exploration Program and Technology Development against the Space Station Freedom accommodations. For Exploration, the program phases support a mission decision for the moon in Fy93 and for Mars in Fy01.

For the lunar mission, Phase A studies continue through Fy92 with Phase B in Fy93 & 94 leading to a Phase C / D in Fy95 and the first manned Lunar flight in FY04. This calls for a major technology / advanced development program over the next 5 years leading to a lunar technology decision in Fy95.

This paper will present a status of the In-Space Transportation System concepts, propulsion concepts,preliminary propulsion design criteria and the technology issues and challenges associated with the next generation In-Space Transportation Systems.

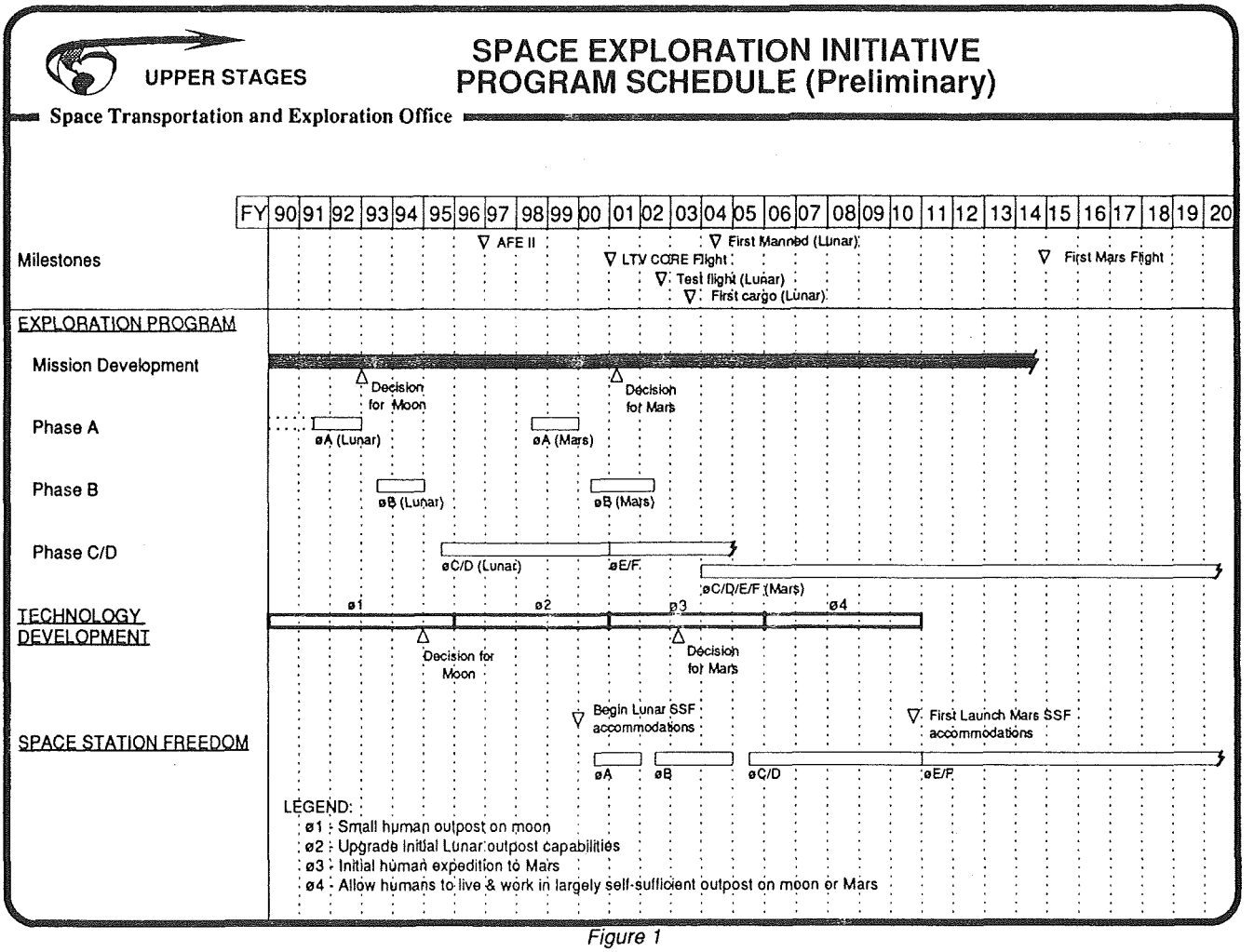


Figure 1

IN-SPACE TRANSPORTATION SYSTEM INTERFACES

The many In-Space Transportation System interfaces are illustrated in Figure #2 and shows other center involvement, proposed and on-going study and technology / advanced development activities and the other Lunar/Mars infrastructure elements i.e. Planetary Surface System (PSS), Earth-To-Orbit (ETO), Space Station Freedom Node for the Space Exploration Initiative (SEI) Program. The Integrated Transportation System activities interface with each and will have a major influence on those specific designs, supporting infrastructure and the overall future technology development program.

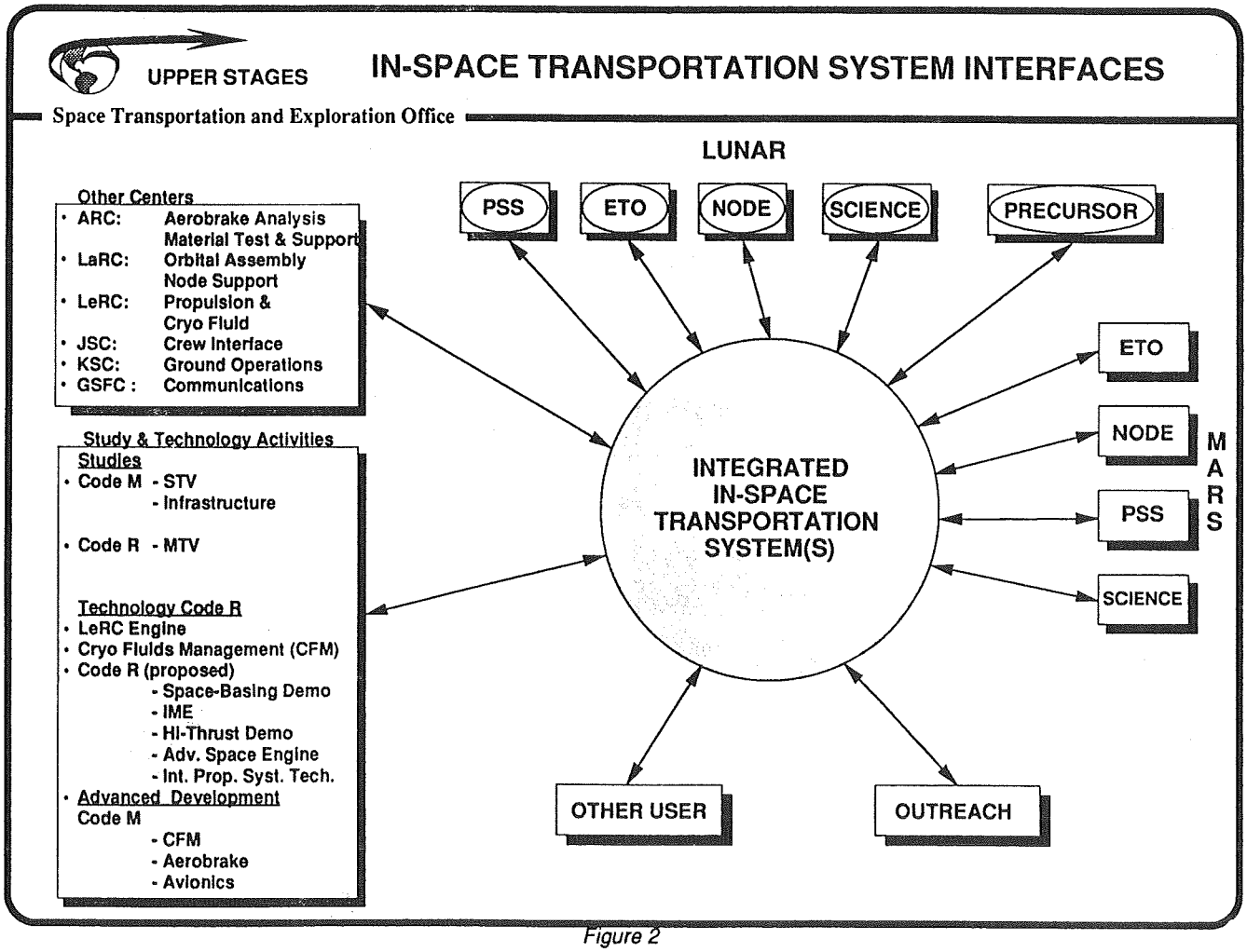


Figure 2

EXPLORATION PROGRAM REQUIREMENTS (OPTION 5)

The top level mission requirements are summarized in Figure #3 for the moon (option 5). (The Mars mission requirements are not presented here.)

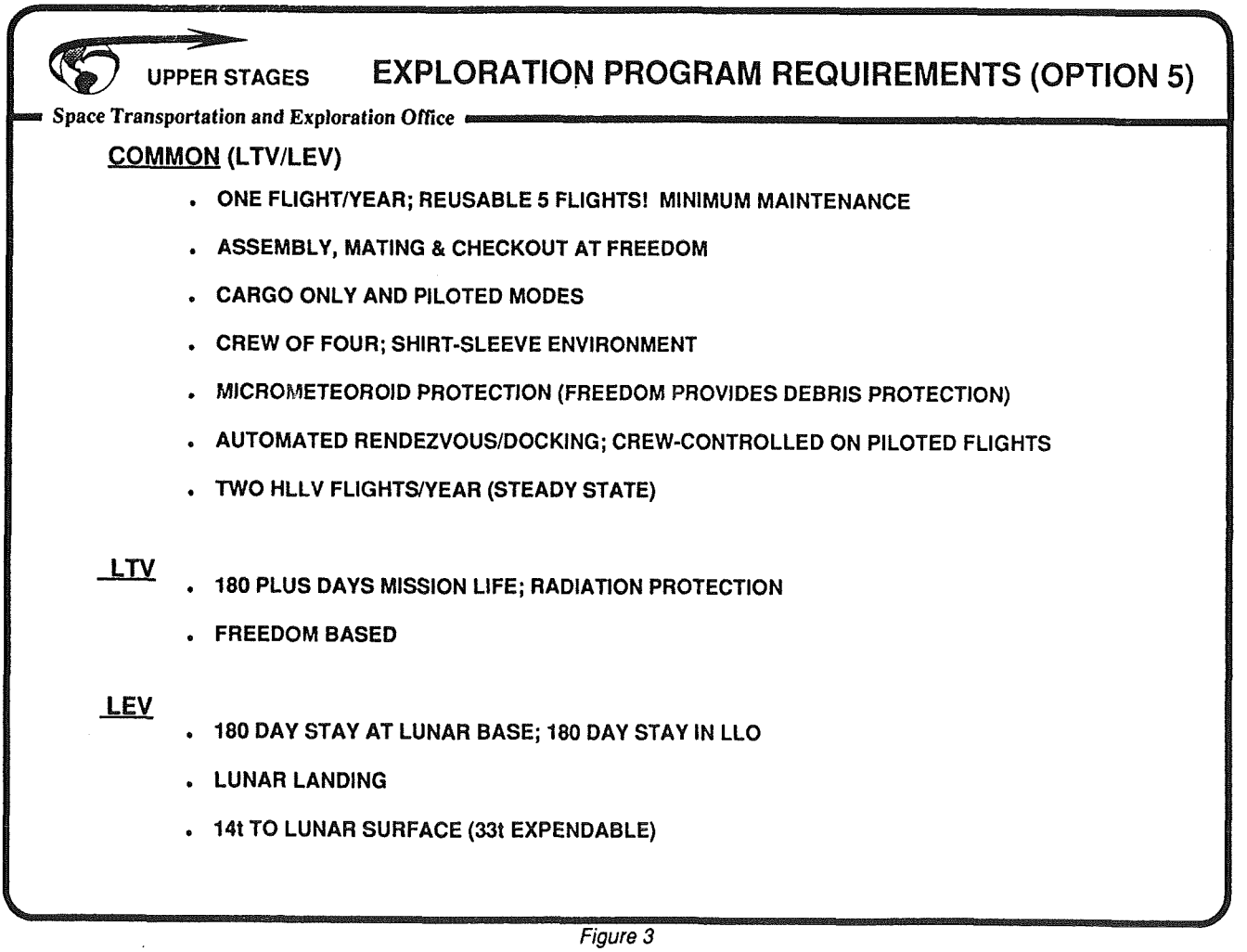


Figure 3

PROGRAM DEVELOPMENT OPTIONS

Shown are three basic program development options for the next generation In-Space Transportation System(s). The technology, timing and the development approach to vehicle evolution are significantly different for each.

The first (I) starts with the simplest and progress to more complicated vehicles. This was the approach of earlier Orbital Transfer Vehicle (OTV) studies. It begins with low technology and evolves with improved technology to the moon and ultimately to the Mars family of vehicles. This is a maximum evolution path incorporating new technology in progressive steps for the moon and later Mars vehicles.

The second (II) is to start with a primary objective of designing for the Lunar transportation requirements and evolve backwards and forwards to satisfy the other missions. Selected high leverage technologies applicable to later Mars missions are emphasized early i.e. propulsion , aerobrake. This is the selected approach under study now as part of the Space Transfer Vehicle (STV) Code M studies. Initial vehicle concepts could include design 'scar' for simpler and earlier expendable missions.

The third (III) is to start with a primary objective of Mars for designing the transportation system(s) and accepting the design impacts for Lunar and other missions. This approach is under study as part of the SEI Code R study.

Each option will be evaluated against and driven by the SEI goals and requirements. The difference is which (if any) is optimized. The challenge is to define what technology and timing best fit each program.

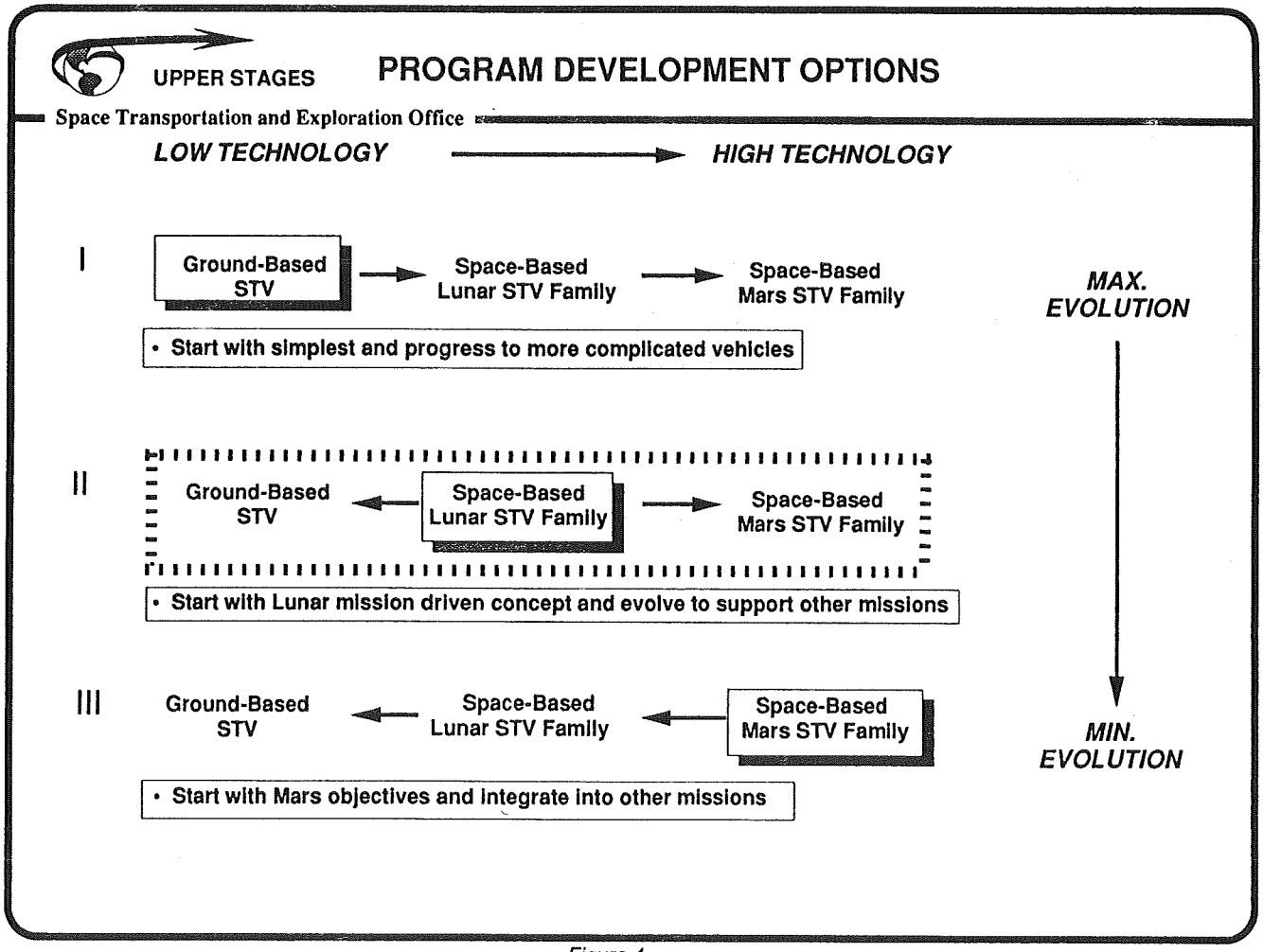


Figure 4

LUNAR TRANSPORTATION SYSTEM (90-DAY STUDY REFERENCE)

The Lunar Transportation System (LTS) was designed, in the 90 day study, to carry 15t of cargo to the lunar surface in the piloted mode and 32t in the cargo (expendable) mode (Figure 5). The LTS consists of the Lunar Transfer Vehicle (LTV) and the Lunar Excursion Vehicle (LEV). The LTV consists of an earth-returnable, reusable core containing a crew module, core systems and an aerobrake, and four (4) propellant tanks that are dropped when expended. The LEV shares common core systems with the LTV and provides the specialized systems for landing and returning cargo from the lunar surface.

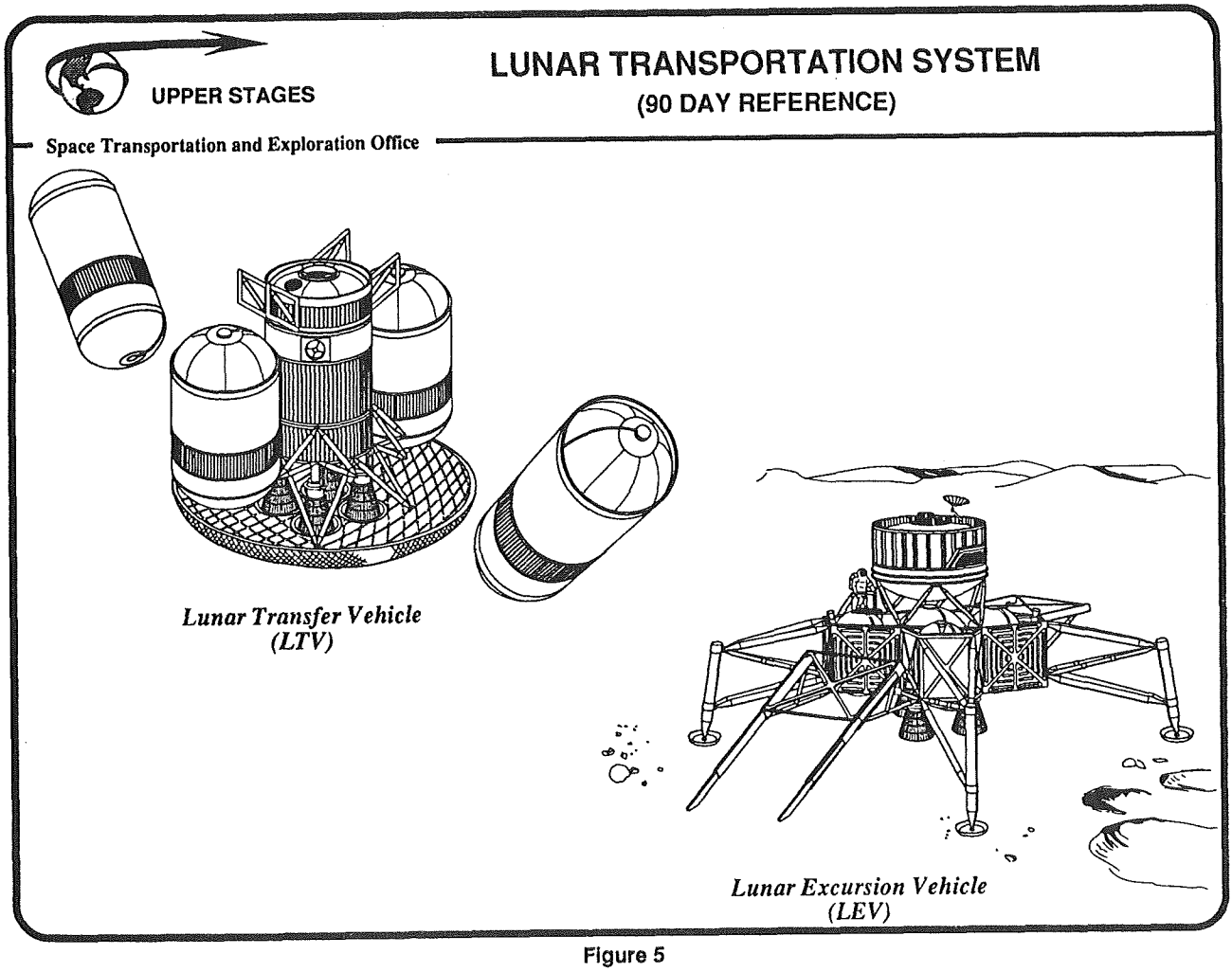


Figure 5

LUNAR TRANSPORTATION OPERATIONS (90-DAY STUDY REFERENCE)

The Lunar Transportation System (Figure 6) is assembled at Space Station Freedom and is launched to the moon by the LTV propulsion system (1). Two TLI tanks are expended after the TLI burn (2). The LEV performs rendezvous and docking with the LTV after it achieves lunar orbit, refuels from the LTV, picks up the arriving cargo and/or exchanges crew (3). The LEV separates from the LTV and delivers the crew and cargo to the lunar surface (4). The two remaining empty tanks are dropped to the lunar surface by the LTV. The LTV then initiates TEI maneuvers and returns to earth orbit for rendezvous and docking with Freedom.

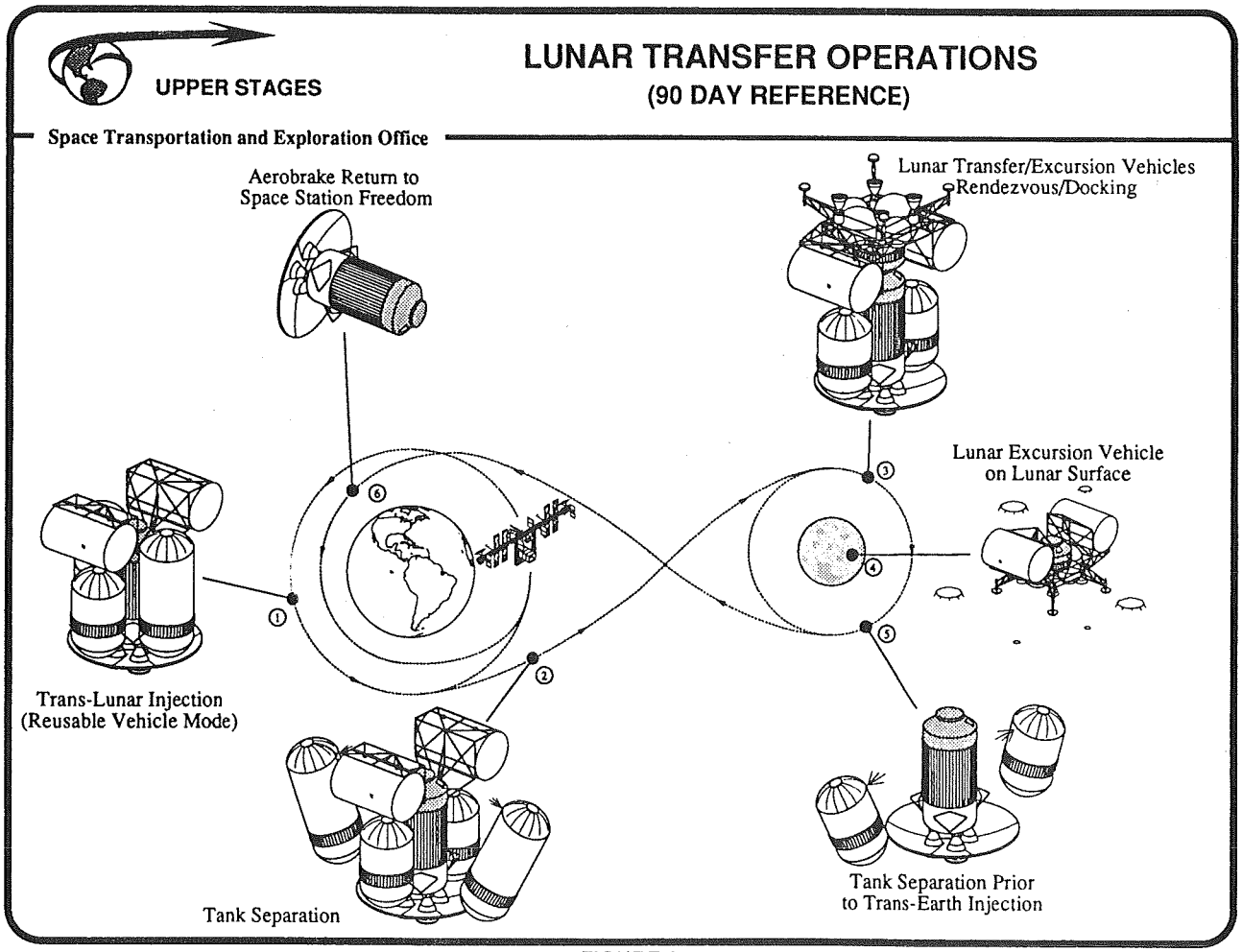


FIGURE 6

PROPULSION CONCEPTS INFLUENCED BY NODE SELECTION

First, the propulsion concept is influenced by the assembly node selected. Chemical propulsion is common to most nodes under consideration and is the most likely propulsion concept for the first decade of the 21st century. As the node location moves further away from earth, the alternative nuclear propulsion concepts become more attractive but, the perceived safety problem of operating a reusable nuclear propulsion system routinely out of and into low earth orbits will be a most difficult problem to overcome.

Second, the sequence and timing of the SEI program as outlined by President Bush (i.e., First Freedom, Return to the Moon and then to Mars) establishes Freedom's availability for the transportation assembly node for Lunar missions. However, on-orbit assembly is only enhancing for Lunar missions but for Mars, on-orbit assembly is enabling. For Lunar we are still evaluating the degree of Freedom use as a transportation node.

Third, Mars Transportation Systems will be enhanced with nuclear propulsion; but chemical propulsion is envisioned as continuing to play a major role in the future Mars missions as well.

Four, major emphasis needs to continue for alternative chemical propulsion technology concepts to best satisfy the Lunar and later Mars missions.

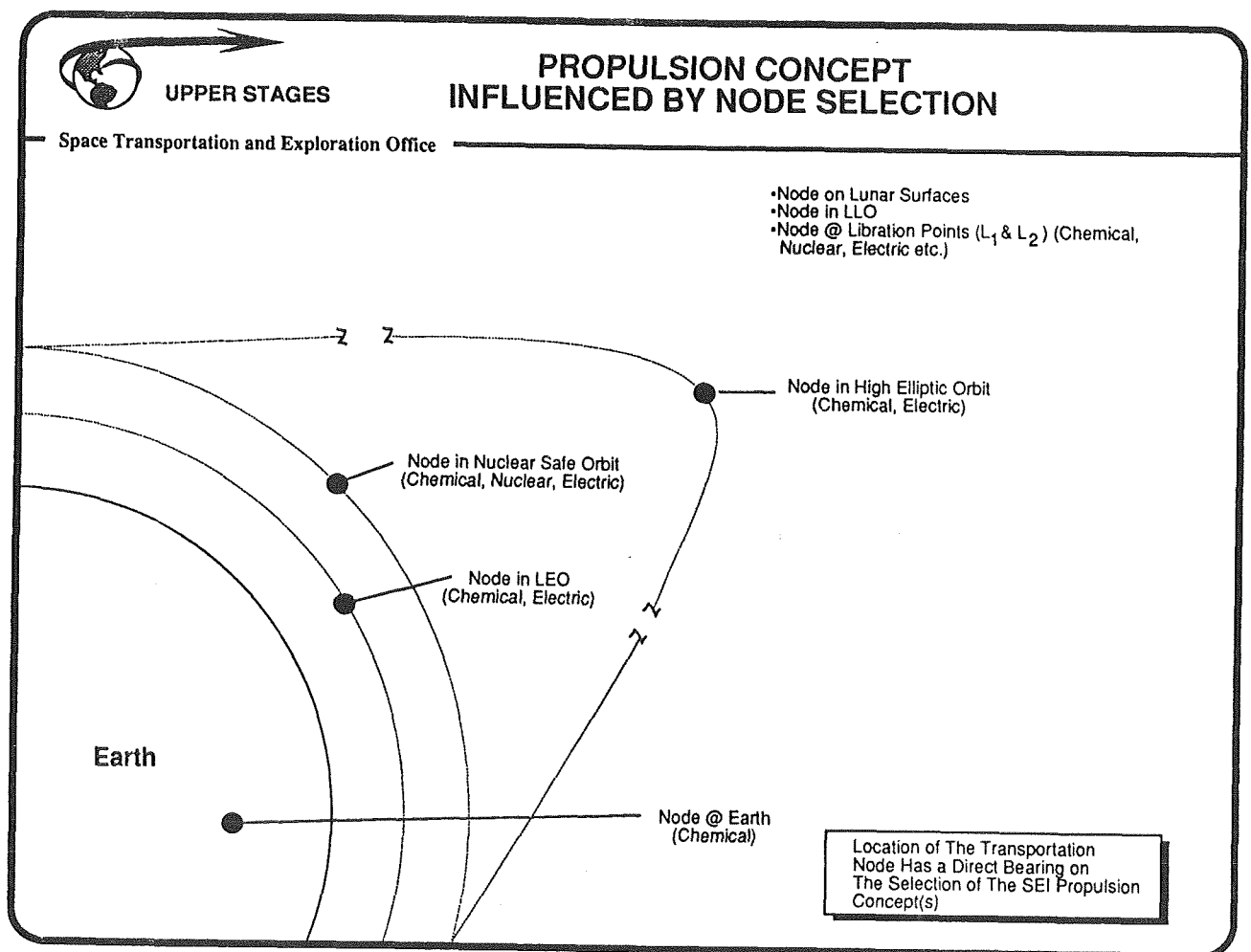


FIGURE 7

PROPULSION CONCEPT APPROACH

Configuration trades for delivering SEI payloads fall into two categories: 1) Common cargo and crew vehicles or 2) Alternatives where the cargo vehicle is the same or different from the crew vehicle.

During the 'in-house' Human Exploration Initiative studies in late 1989, our approach placed emphasis on commonality for the individual mission requirements for cargo and crew on the same vehicle. Since then we have expanded to identify and conceive alternative conceptual configurations for cargo, combined and crew only missions to meet the Lunar, near earth, planetary delivery and Mars exploration requirements.

Figure 8 illustrates four propulsion types and several vehicle architectures and reusable options, with different evolutionary implications. Our current contractor studies are focusing on chemical (LO2/LH2) propulsion systems for the Lunar, mission earth, non-NASA, precursor, etc., mission requirements and applications for Mars Missions. (Other on-going studies are looking at broader alternative concepts i.e., nuclear, solar, electric, etc.)

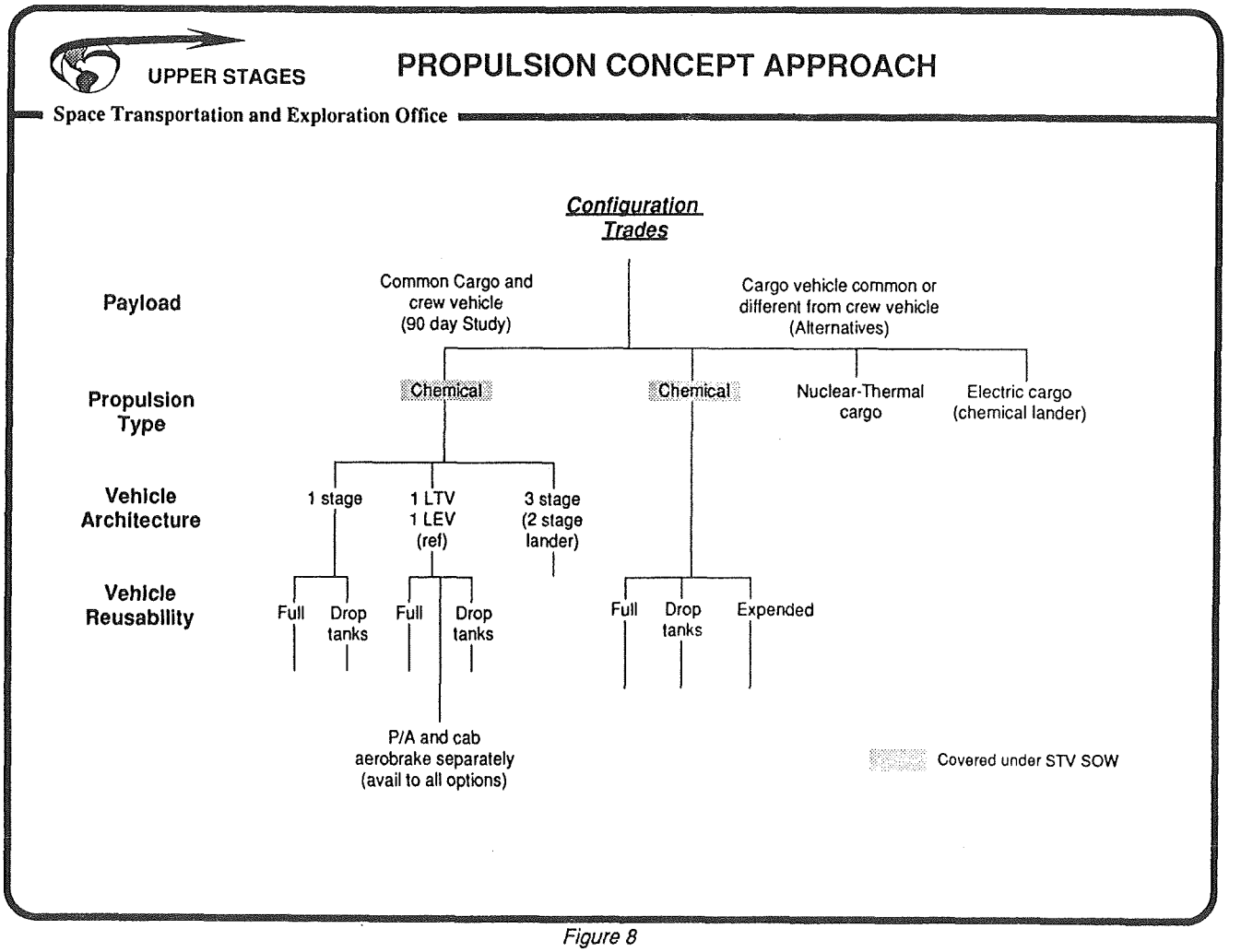


Figure 8

Key Development Criteria for Next Generation Engine

The SEI program provides the opportunity to begin evaluating key development criteria for the next generation space engine. A preliminary list of engine criteria is shown on Figure 9. These criteria are presented in two parts: 1) Generic (all missions) and 2) Specific (mission dependent).

Reliable
High reliability is essential for dependable vehicle operations and safety for all missions. Reliability may be obtained by: redundancy or "robust" design or combinations of both, by an exhaustive test program or by improved subsystem component and interfacing reliability i.e., health monitoring sensors.

Space-Based
Space-basing is necessary for permanent human exploration missions and is based on:
1) the need for on-orbit assembly of the large Lunar or Mars Integrated Transportation System
2) reusability
3) the need for routine transportation to establish permanent human presence beyond earths orbit.

From the vehicle viewpoint, the Integrated Lunar or Mars Transportation systems are large enough that final assembly must be accomplished in orbit. This will require the capability to mate, de-mate, inspect, test, refurbish, and maintain the vehicle before, during and after a mission.

From an engine viewpoint, the space-based engine will be designed for minimum maintenance, have a comprehensive health monitoring system utilized for pre-mission checkout, realtime safety monitoring and incipient failure mode identification, post-firing trend monitoring; and will be designed to withstand long exposures to the space environment.

Man-Rated
Man-rating is the process of evaluation and assuring that the hardware and software can meet prescribed, safety-oriented design and operational criteria. It is an integral part of the design, development, verification, management and control process and encompasses the complete design concept from Phase A to Phase E and F. It is characterized by: high reliability, failure tolerance, design and installation for contained damage, design or processing changes in response to failures, comprehensive test programs and crew interaction. Redundant components or engine-out capability may be required. The crew will be provided with fault detection, isolation and reconfiguration capability of critical systems.

Long-Life
As a goal, the vehicle / engine will be designed for five years or five mission life while exposed to the space environment. System / subsystem degradation must be incorporated into the design factors of the Integrated Transportation System. Material selection and development for space-based engines may emerge as an important design criteria after examination of the available data (particularly that from the Long Duration Exposure Facility, LDEF).

Key Development Criteria for Next Generation Engine (Continued)

Engine Throttling
Engine throttling is necessary for accurate and safe landing. Throttling operations will require extensive study. It is not clear how fast the engine must respond to throttling requirements nor whether the engine must operate continuously over the full range or can pass through some ranges in a transient manner.

Vehicle / Engine Interface
Interfaces must be simple and reliable, commensurate with the space-basing requirement, but are otherwise subject to vehicle / engine trades. For example, turbopumps and combustion chambers might be manifolded for redundancy. This will place major emphasis on control, health monitoring and reliable diagnostic sensors.

Health Monitoring
A good health monitoring system capable of preflight, flight and post flight diagnostics, fault isolation, and safety monitoring is essential for a man rated, space-based engine. Whether the system is best lodged in a central vehicle data processing system, a propulsion system data processing system, or engine-mounted controller is not clear. Redundant data processing and storage may be desirable for some or all of the engine data.

Margins
High reliability will require a robust design, insensitive to operational conditions at the design point. Margins must be demonstrated by test, although this may not require test to destruction.

Performance
All performance specifications, including thrust, specific impulse (Isp), and mixture ratio are subject to trades. Size specification, primarily driven by gimbal angle requirements and fixed vehicle diameter, will influence the chamber pressure selection, expansion ratio, and Isp selection. The engine will probably be required to operate over some range of mixture ratios for efficient propellant utilization.

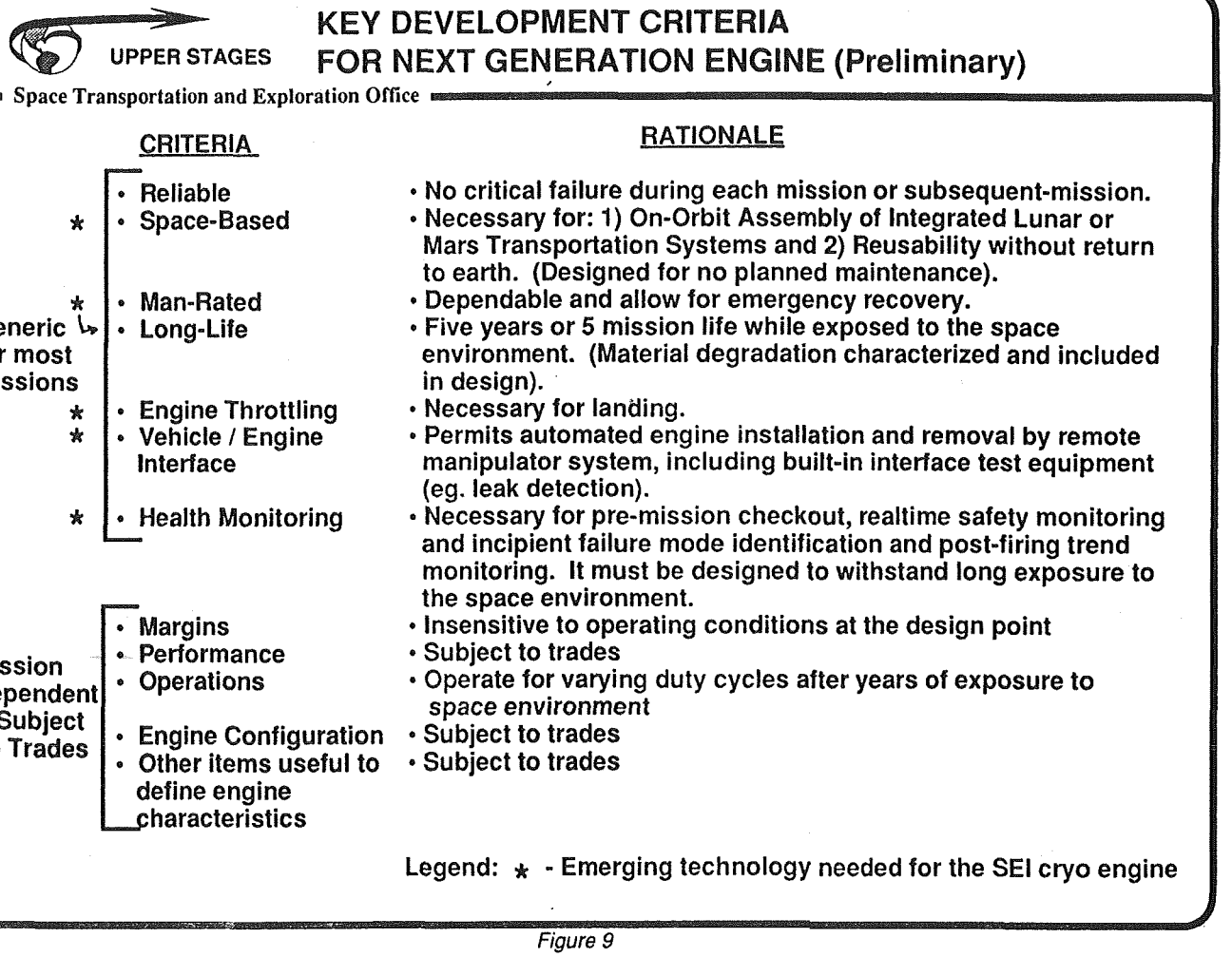

UPPER STAGES
KEY DEVELOPMENT CRITERIA
FOR NEXT GENERATION ENGINE (Preliminary)
Space Transportation and Exploration Office
CRITERIA
RATIONALE
Generic for most Missions
Mission Dependent & Subject To Trades
• Reliable
• Space-Based
• Man-Rated
• Long-Life
• Engine Throttling
• Vehicle / Engine Interface
• Health Monitoring
• Margins
• Performance
• Operations
• Engine Configuration
• Other items useful to define engine characteristics
• No critical failure during each mission or subsequent-mission.
• Necessary for: 1) On-Orbit Assembly of Integrated Lunar or Mars Transportation Systems and 2) Reusability without return to earth. (Designed for no planned maintenance).
• Dependable and allow for emergency recovery.
• Five years or 5 mission life while exposed to the space environment. (Material degradation characterized and included in design).
• Necessary for landing.
• Permits automated engine installation and removal by remote manipulator system, including built-in interface test equipment (eg. leak detection).
• Necessary for pre-mission checkout, realtime safety monitoring and incipient failure mode identification and post-firing trend monitoring. It must be designed to withstand long exposure to the space environment.
• Insensitive to operating conditions at the design point
• Subject to trades
• Operate for varying duty cycles after years of exposure to space environment
• Subject to trades
• Subject to trades
Legend: * - Emerging technology needed for the SEI cryo engine

Figure 9

EMERGING VEHICLE / ENGINE TECHNOLOGIES

Five technology challenges are emerging as a result of the vehicle / engine criteria development workshop and the STV Concepts and Requirements Studies currently underway. These technologies are:

- Space basing
- Long-life (material development and processes)
- Engine Throttling
- Vehicle / Engine Interface
- Health monitoring and control

Although an explicit technology requirement cannot be identified at the present, technologies that provide "robustness" and minimize health monitoring system reliability are sought. This is a high payoff area of sensor technology.

Other technology areas are expected to emerge as the vehicle, engine and vehicle / engine interface definition emerge. These development criteria and emerging technology reflect the need to reduce the cost of launching and expending hardware and to establish a high confidence that later, longer duration missions to Mars will be successful.

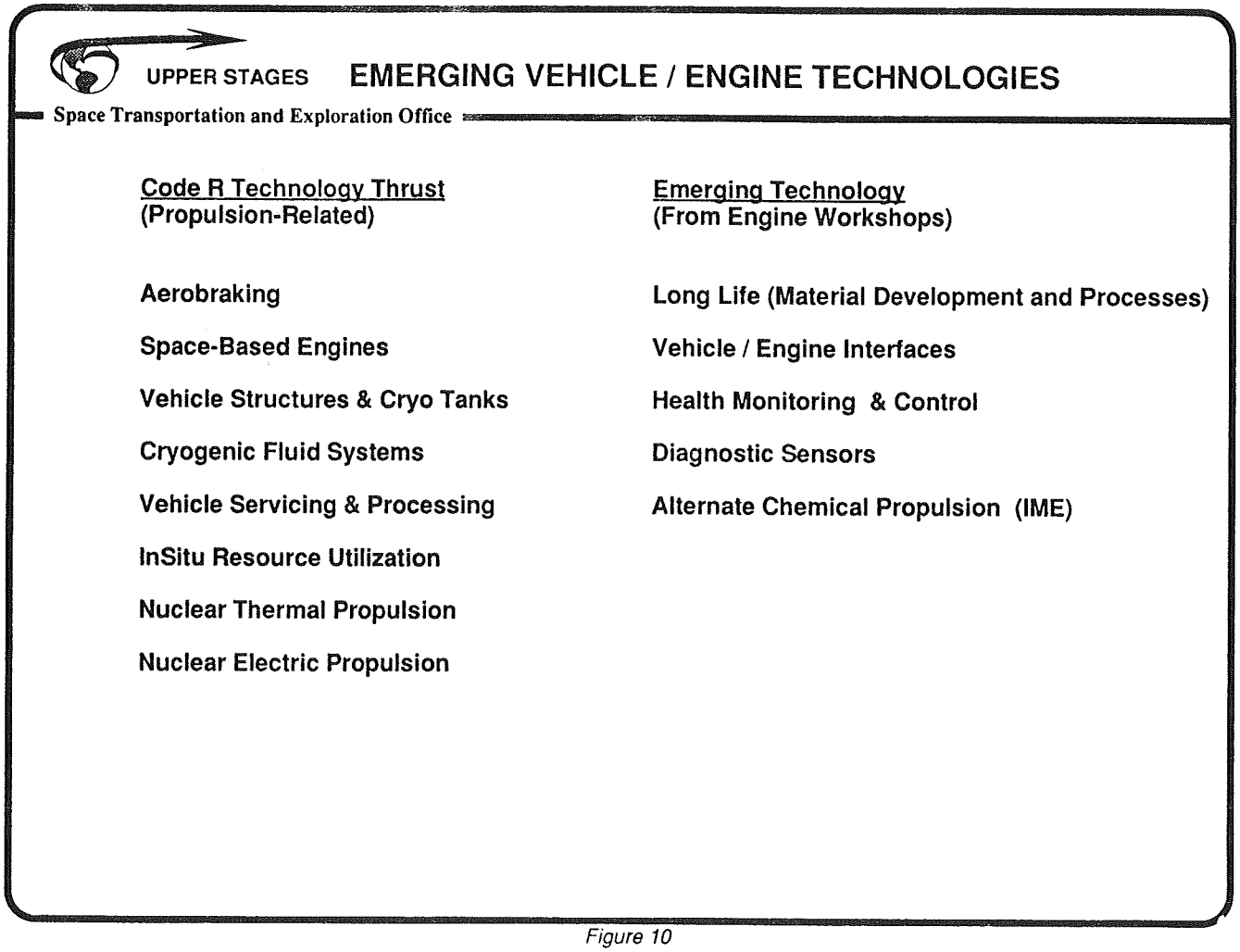


Figure 10

SPACE-BASED ENGINE TECHNOLOGY ISSUES

Vehicle/Engine

The approach to Man-Rating should foscus on increasing reliability of components, manufacturing processes, and sensor technology. The current systems should be improved before we start adding additional" bells and whistles." Extensive health monitoring should not be used as a substitute for high design margins. Increasing sensor reliability is critical to improving health monitoring. Design engine with margin for growth in thrust. Thrust growth should be anticipated before development and then that value must not be exceeded. Performance, operation and structural margins should be established for maximum anticipated vehicle demand and should be maintained during adjustments to mission requirements. That is, margins should never erode below normal levels to meet future growth requirements.

Nozzle

Work is needed to develop light weight high reliability nozzle extensions. Extendable/retractable nozzles will be required for launch packaging, aerobrake maneuvers, and landing of some transportation concepts under consideration.

Turbo Pump Assemblies (LH2 and LOX)

Technology work in the turbo pump area is critical to the success of the Advanced Space Engine. Low NPSP pumps are needed to have the most efficient vehicle design because inlet pressure can drive tank design. These pumps will need to be lightweight, compact, and very reliable to meet the mission requirements. The pumps will need to be designed for long life (5 years) with many restarts (100) and operate over a wide throttling range (20:1).

Combustor

Technology work is needed on the Thrust Chamber Assembly (TCA) because of the unique operating environment the ASE will be subjected to; the ASE must be capable of starting on gas/gas, gas/liquid or liquid/liquid. It must be capable of long life with many restarts, high efficiency with throttling, high reliability, and have low ΔP injectors for tank head starts. With the expander cycle design and high chamber pressures which require technology work.

The ASE engine will be very beneficial to the SEI program and as described here requires a concentrated technology effort to realize the goal of an efficient, highly reliable engine which can contribute to the LTS and MTS vehicles.

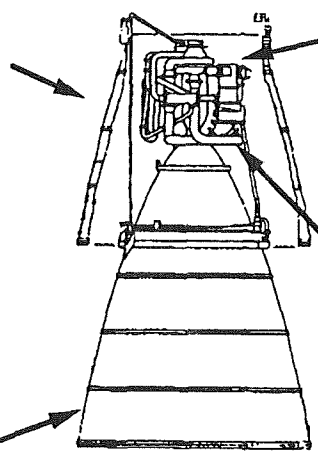
UPPER STAGES

SPACE-BASED ENGINE TECHNOLOGY ISSUES

Space Transportation and Exploration Office

VEHICLE/ENGINE

- Space Basing
- Man Rating Approach
- Reusability/Long Life
- Health Monitoring
- Materials



TURBO PUMP ASSEMBLIES (LH2 & LOX)

- Low NPSP Pumps
- Lightweight/High Efficiency Pumps
- Turbine Drive Cycle (Expander, Gas Generator, Staged Comb.)
- High Reliability
- Health Monitoring
- Long Life w/Many Restarts
- Space Maintenance
- Throttling

COMBUSTION

- Health Monitoring
- Long Life w/Many Restarts
- Space Maintenance
- Throttling
- High Combustion Efficiency
- Low Injector ΔP
- High Chamber Pressure (900-1500 psia)
- Low-Cost Chamber Materials and Manufacturing Process
- Robust Chamber/Nozzle Regenerative Cooling Method

NOZZLE

- High Expansion Ratio (Up to 1000:1)
- Low Weight Nozzle Extension
- Lightweight Deployable 2-Position Nozzle w/High Reliability
- High Isp (up to 490 Sec)

Figure 11

TECHNOLOGY / ADVANCED DEVELOPMENT-PROPOSED ACTIVITIES

A program to define and demonstrate an Advanced Space-Based Engine for future in-space transportation applications is in progress at LeRC. MSFC is proposing a space-based engine project to identify and demonstrate the technology necessary for space-basing modifications to existing technology necessary for space-basing modifications to existing low and high thrust engines (RL-10 and J2/J2S) and to investigate the feasibility of development of integrated Modular Engine. These activities are being coordinated through engine workshops at MSFC. These workshops are bringing together the vehicle and engine communities in a common forum to develop a coordinated set of engine requirements. The workshops are also serving to focus the approaches to integrated Propulsion System technologies in areas such as space-basing, philosophy, monitoring and control, integrated sub-systems approaches, advanced mechanisms, power systems approaches and requirements analysis approaches. LeRC and MSFC are also cooperating in the development of technologies for Cryogenic Fluid Management. The CFM projects are developing test-beds for cryo-tank hydrogen demonstrations at MSFC and for flight demonstrations of cryogenic transfer by LeRC (COLDSTAT).

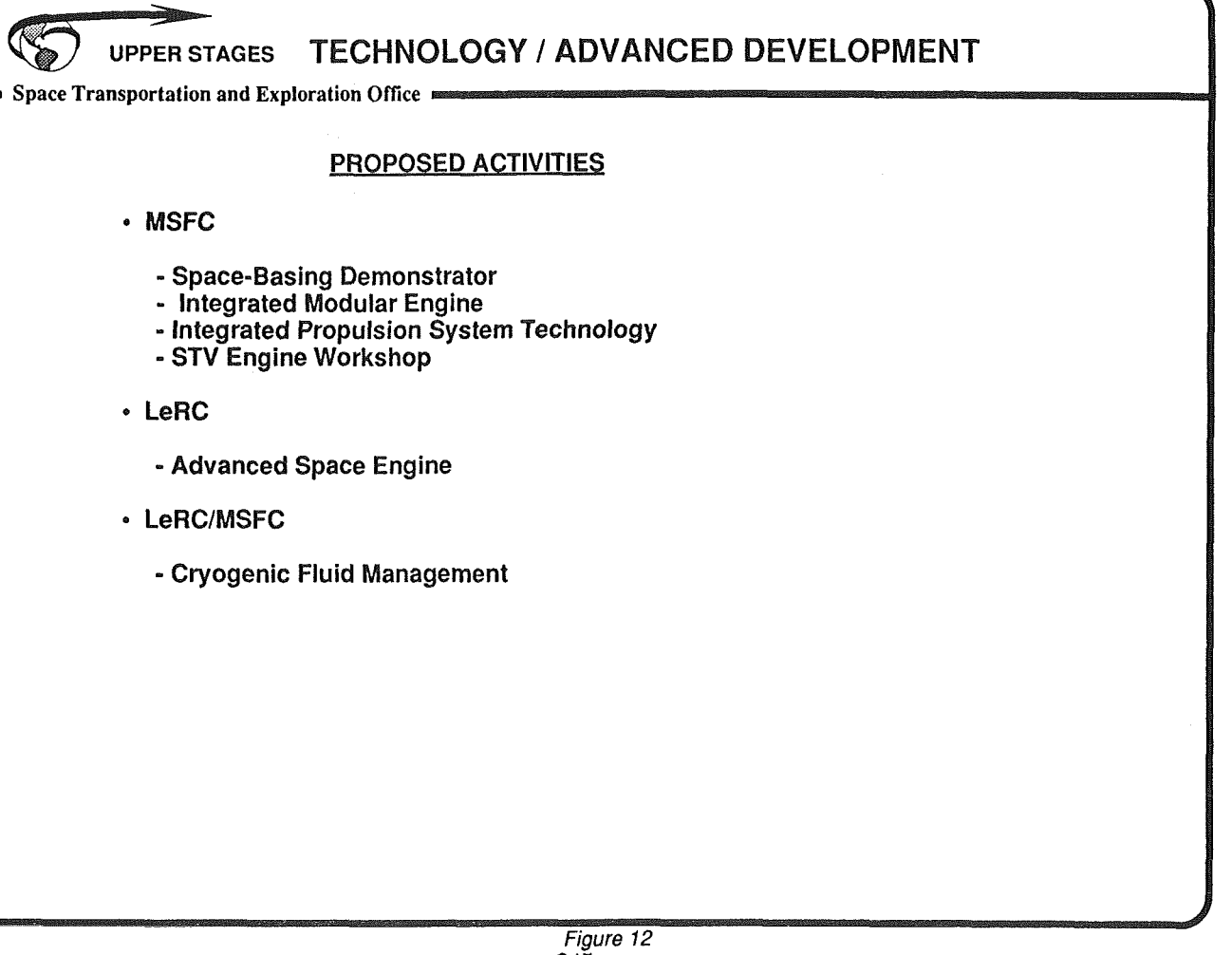


Figure 12

SPACE-BASING DEMONSTRATOR

This project will investigate the modifications necessary to update an RL-10 engine to space-basing requirements and demonstrate those capabilities in ground tests. Specific areas of investigation include sensors and data processing for automated pre-mission checkout, real-time failure monitoring and failure mode identification, post-firing trend monitoring, modifications to permit automated installation, purge elimination and reduction of non-propulsive consumables.



UPPER STAGES

SPACE-BASING DEMONSTRATOR

Space Transportation and Exploration Office

GOAL:
- Identify and demonstrate the technology necessary for "Space-Basing"

APPROACH:
- (Based on the RL-10 expander cycle engine)
- Identify the sensors and data processing required to automate the pre-mission checkout, realtime safety monitoring and failure mode identification, and post-firing trend monitoring
- Identify modification to the vehicle interfaces to permit automated installation
- Develop a test plan to minimize or eliminate consumable non-propulsive fluids
- Implement the health monitoring system, modified vehicle interfaces and any minor modifications that would eliminate purges on an L-10
- Install the engine in a test stand employing the "space-based" interface. Do all necessary checkouts remotely, fire the engine, do post-firing hardware and data evaluation and repeat TBD times
- Investigate reduction in non-propulsive consumables per test plan

BENEFITS:
- Using an existing expander cycle engine gives access to known FMEA/CIL, hazards analyses, and operating procedures on which to base the health monitoring effort
- RL-10 has demonstrated benign failure behavior in case of errors in analyses
- Use of existing engine permits rapid completion of technology program
- RL-10 represenmts an inexpensive testbed for space-basing technology
- The focus on changes to an existing system assures identification of minimum technology requirements

Figure 13

INTEGRATED MODULAR ENGINE

The need to provide an engine with variable length and thrust, new strategies for redundancy and safety and one that is capable of being integrated with an aerobrake has fostered a new engine approach: the Integrated Modular Engine. This engine project will provide the technology necessary to confidently proceed in the 1990's with the development of a modular plug-nozzle liquid hydrogen oxygen expander cycle engine for future space exploration missions. This proof-of-concept test bed will validate design analysis methodologies and define optimum requirements and characteristics for the engine.

UPPER STAGES

INTEGRATED MODULAR ENGINE (IME)

Space Transportation and Exploration Office

Objective:

Develop and demonstrate the technology necessary to apply modular plug-nozzle engines for future space exploration missions.

System studies to identify optimum engine system architecture

- Redundancy Management
- Differential Throttling
- Throttling Range

Approach:

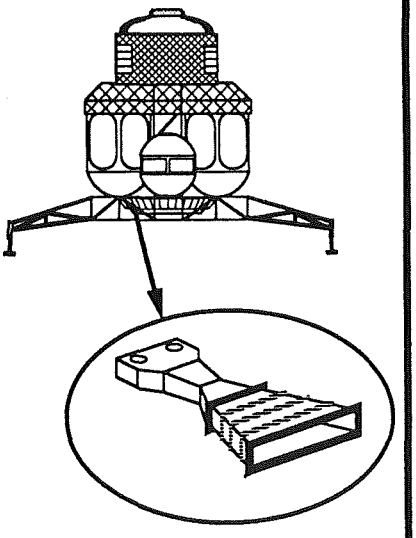
Establish best approach to multiple chamber vacuum ignitor

Component technology development

- Throttleable Injectors
- Turbomachinery
- Utilize past work and LeRC work in area where possible

Design, develop and assemble IME breadboard

Test



Benefits:

Provide the technology base for an alternative engine system design which may integrate better with the space exploration vehicles and have improved system reliability compared with the baseline bell-nozzle engines

Figure 14

CRYOGENIC FLUID MANAGEMENT

The handling of cryogenic fluids in low-g conditions poses many problems that affect how we operate space-based transportation vehicles i.e., engine start sequences, propellant settling, quantity gaging, tank fill and refill, tank chilldown, fluid transfer and residuals disposal. Operational work-arounds have been defined to resolve most of these problems with attendant operational inefficiencies. LeRC and MSFC are cooperating on a series of flight and ground tests to characterize CFM technology approaches and develop the procedures, operations, controls and instrumentation to enable design solutions for cryogenic fluid handling in future space transportation vehicles. These experimentation programs will develop the analytical models needed to define the parametrics in support of future propulsion system design.



UPPER STAGES **CRYOGENIC FLUID MANAGEMENT**

Space Transportation and Exploration Office

Objective: Define and develop experimentation to investigate the technologies for managing propulsion cryogenic fluids in space.

Approach:

- Systems studies to define requirements for cryogenic fluid management.
- Definition of ground tests to investigate propulsion cryogenic fluid management technologies such as no vent fill, start baskets, venting approaches, insulation techniques, quantity gaging and residuals disposal.
- Scaling data from 1-g tests to support extrapolation from LH_2 to LN_2 with simulation of LO_2 results.
- 1-g Analytical Models.
- Develop operations, procedures, controls and instrumentation.
- Flight tests support scaling of 1-g models to low-g conditions.

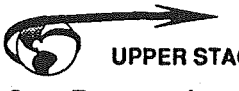
Benefits:

- Reduction of operational work-arounds (i.e., acceleration settling of fluids)
- Simple vehicle operations; fluid disposal, engine start sequences, quantity gaging, fill/refill, tank chilldown and fluid transfer.

Figure 15

INTEGRATED PROPULSION SYSTEM TECHNOLOGY

The development of requirements for the transportation vehicle and its engines must provide an architecture that provides coordinated design approaches for common systems such as power and fluids. Coordination of common systems requirements and the implementation of these requirements in vehicle or propulsion systems that most efficiently serves the integrated transportation systems needs will reduce overall vehicle weight and redundant overlaps in system design. Potential design efficiencies in areas such as power generation, system monitoring and control, integrated cryo fluids systems and advanced mechanisms and the promise of reduced cost and weight warrant further efforts in this area.



UPPER STAGES

INTEGRATED PROPULSION SYSTEM TECHNOLOGY

Space Transportation and Exploration Office

Goal: Define the requirements and technologies necessary to combine the vehicle and engines in an integrated propulsion system.

Approach: System studies to identify architectures for the vehicle and engines that will incorporate integrated design approaches in the following areas:

- Space Basing Requirements/Philosophy
- Propulsion System Monitoring and Control
- Integrated Cryogenic Subsystems
- Aerobrake Aperture/Closure
- Advanced Mechanisms
- Power for Propulsion Component (and Vehicle)
- Vehicle and Mission Requirements Analysis

Benefits:

- Integrated Vehicle/Propulsion design approaches
- Design efficiencies resulting in reduced weight and consumables
- Common development philosophies and lower cost

Figure 16

STV ENGINE WORKSHOP

The NASA, vehicle systems contractors and engine contractors have met in April, 1990 to help focus the propulsion engine technology/advanced development program and also to help define criteria and trades with which to select the overall approach for the Space Transfer Vehicle propulsion systems. A preliminary set of top level criteria was defined for a space-based engine and key trades were identified. Another session of the workshop is planned for early August. The proposed topics are shown in Figure 17.



UPPER STAGES **STV ENGINE WORKSHOP**

Space Transportation and Exploration Office

OBJECTIVE: Primary Emphasis -

- Help focus the propulsion engine technology programs
- Help select the STV concept
- Provide data for contractor and in house government studies
- Define the approach to the development of engine design criteria for the National Space Transfer Vehicle or family of vehicles

ORGANIZATION:

Chairman:	Fred Huffaker, MSFC/Program Development	Vehicle Contractors:	Boeing Aerospace
Co-Chairman:	Jerry Redus, MSFC/Propulsion Laboratory		Martin Marietta
NASA Centers:	NASA Headquarters	Engine Contractors:	Aerojet
	Lewis Research Center		Pratt & Whitney
	Kennedy Space Center		Rocketdyne
	Jet Propulsion Laboratory		
	Johnson Space Center		
	Marshall Space Flight Center		
	Stennis Space Center		

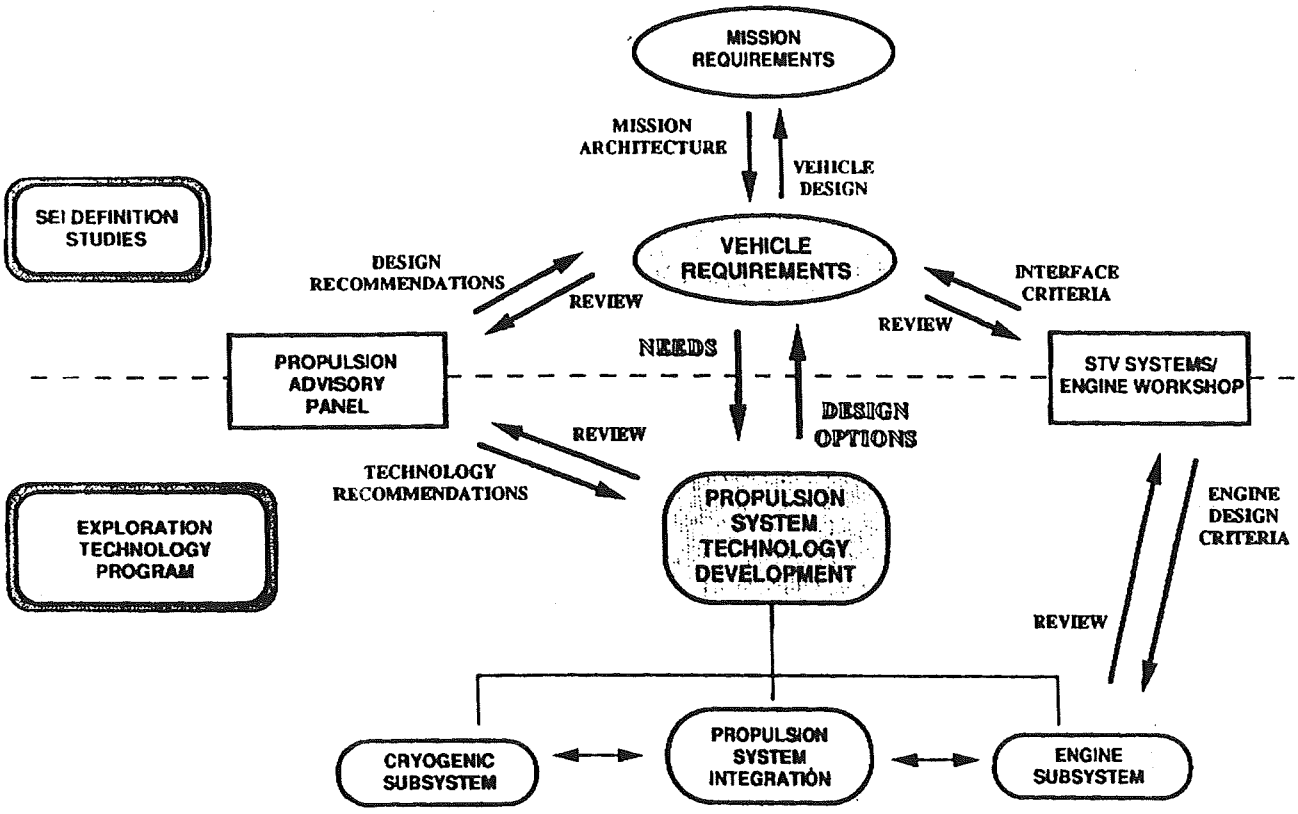
PLANNED TOPICS AT SECOND WORKSHOP (MID-JULY, 1990)

Directed toward a limited number of key issues

- Space Environment and its effect on long-term use of materials as it affects engine life
- Operations (including mainline, maintenance, and contingency)
- Man-rating (demonstration technology and health monitoring)

Figure 17

ETP CHEMICAL TRANSFER PROPULSION



PROGRAM CHALLENGE/SUMMARY

The development of the Next Generation Space Transportation Propulsion System presents a challenge unique to the aerospace program. Never before have the requirements for long life, space-basing, reliability, man-rating and minimum maintenance come together in one transportation system that must protect the lives of our space travelers, support the planetary logistic needs and do so at a reasonable cost. The use of robotics presents additional challenges and solutions to the propulsion designer. This presentation has put the issues "on the table" that must be resolved by the propulsion community. Proposed activities to define the vehicle system requirements and define engine test beds are supportive of future plans to conduct an integrated government/contractor propulsion development program.



UPPER STAGES

PROGRAM CHALLENGE / SUMMARY

Space Transportation and Exploration Office

CHALLENGE:

- Develop a Next Generation Space Transportation Propulsion System for the manned exploration of the planets
- Develop a long life, space - based propulsion system that requires minimum maintenance, is reusable and is capable of repair with the use of robotics

SUMMARY:

- Preliminary definition of the system requirements in progress
- Definition of preliminary propulsion criteria development process / trades / technology (planning group)
- Issues "on the table" to challenge engine community
- Testbeds defined and in work to identify and define engine characteristics (LeRC and MSFC)
- Plans in progress to define and conduct an integrated government / contractor engine development program

Figure 19

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ADVANCED CRYO PROPULSION SYSTEMS

by William K. Tabata

NASA Lewis Research Center

Presented at

NASA/Pennsylvania State University

Transportation Propulsion Technology Symposium

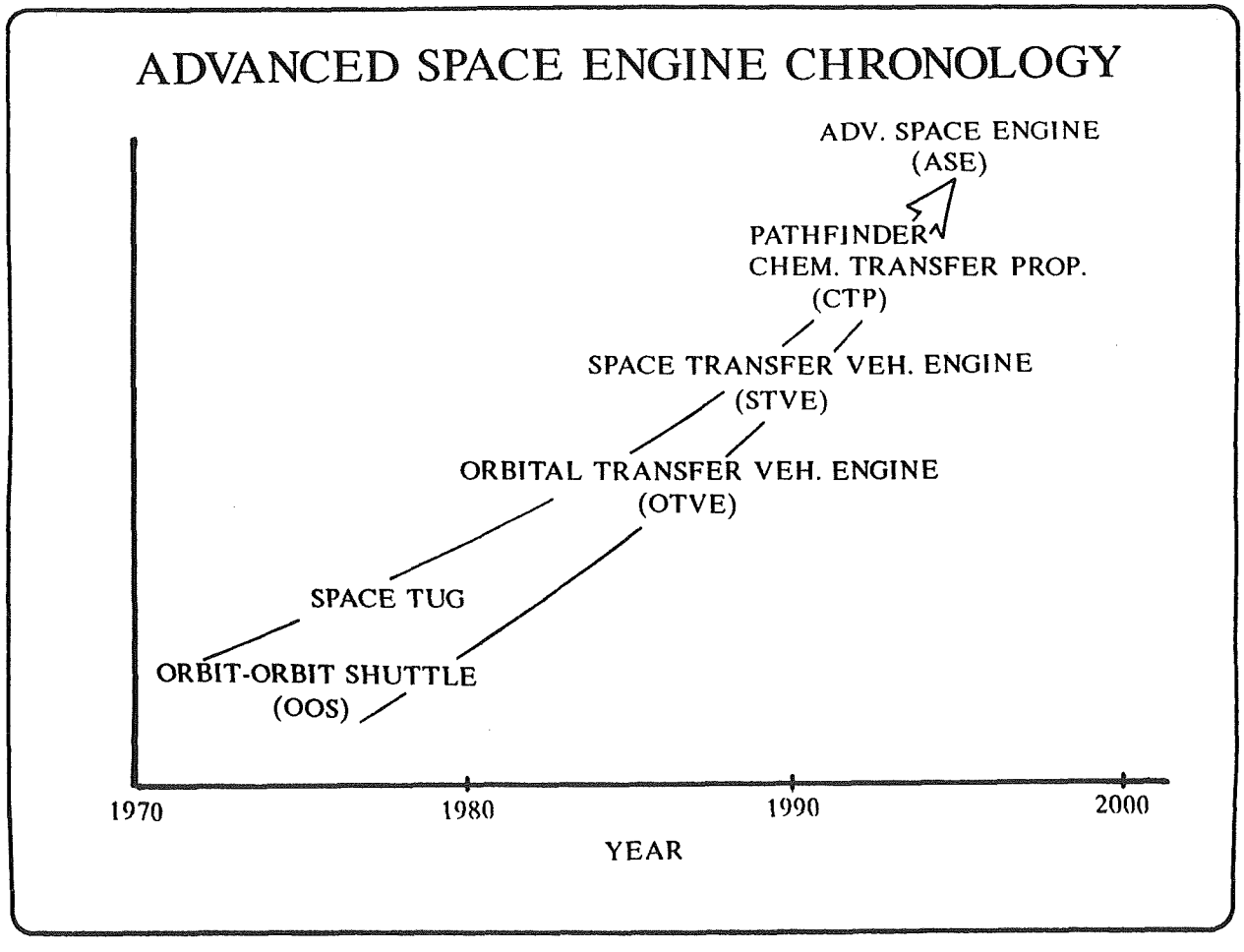
June 25-29, 1990

Lewis Research Center

NASA

An advanced rocket engine for space application has had a long history. Studies started in the late 1960's and early 1970's for an Orbit-to-Orbit Shuttle (OOS) and have progressed through the years to the current Pathfinder Chemical Transfer Propulsion (CTP) Program. Starting in 1991, the CTP Program will be re-titled the Advanced Space Engine (ASE).

During the studies, various propellant combinations and engine cycles have been evaluated. The propellant combination selected is liquid hydrogen and liquid oxygen. The engine cycle selected is the expander cycle because of its simplicity, potential long-life, and high performance.



The new space engine must not only have high performance at rated thrust, but must also be capable of deep-throttling and capable of idle mode operation. The engine must also be man-rated, reuseable, space-baseable, and fault-tolerant.

ASE DESCRIPTION

PROPELLANTS:	HYDROGEN/OXYGEN
CYCLE:	EXPANDER
THRUST:	7.5K to 50K
THROTTLING:	20:1 MAXIMUM
IDLE MODES:	TANKHEAD (NON-ROTATING)
	PUMPED (LOW NPSH INLET CONDITIONS)
REUSE:	LIFE >5 MISSIONS
MAN-RATED:	
SPACE-BASED:	LONG SPACE EXPOSURE
	NO PLANNED MAINTENANCE
	MINIMAL CHECKOUT
	HEALTH MONITORING

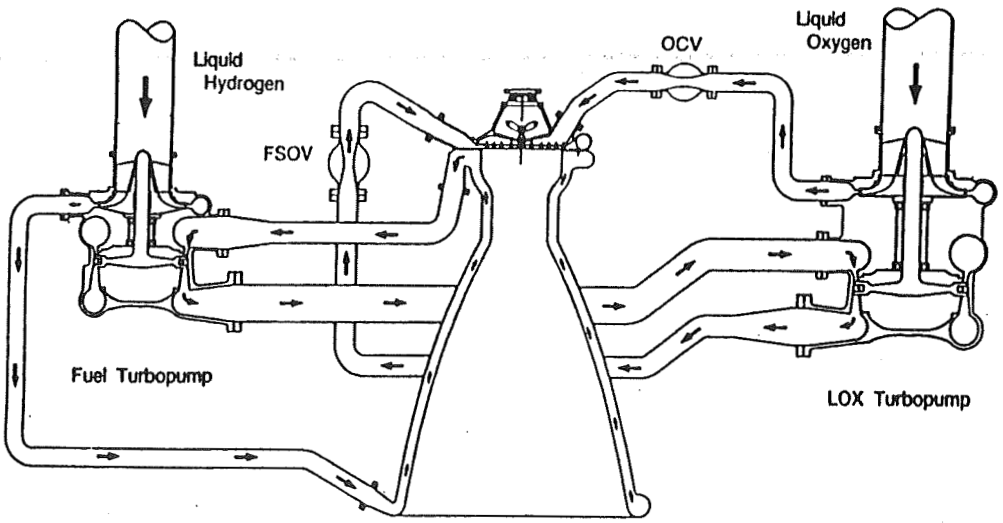
ASE DESCRIPTION (CONT.)

FAULT-TOLERANT:

- BENIGN FAILURE MODES
- HEALTH MONITORING
- SYSTEM CHARACTERIZATION
- ARTIFICIAL INTELLIGENCE

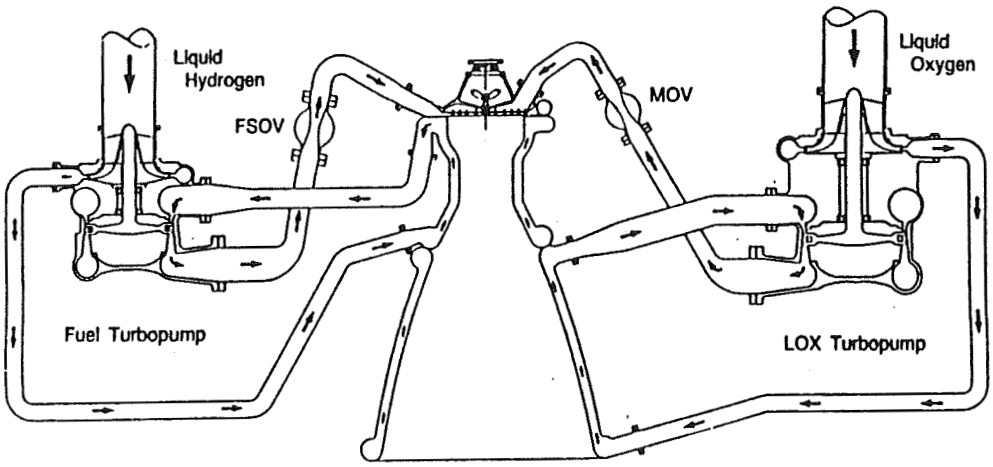
There are three basic expander cycles being evaluated. The SINGLE EXPANDER uses one propellant, the hydrogen, to cool the combustion chamber and provide energy to drive the turbopump turbine(s).

SINGLE EXPANDER



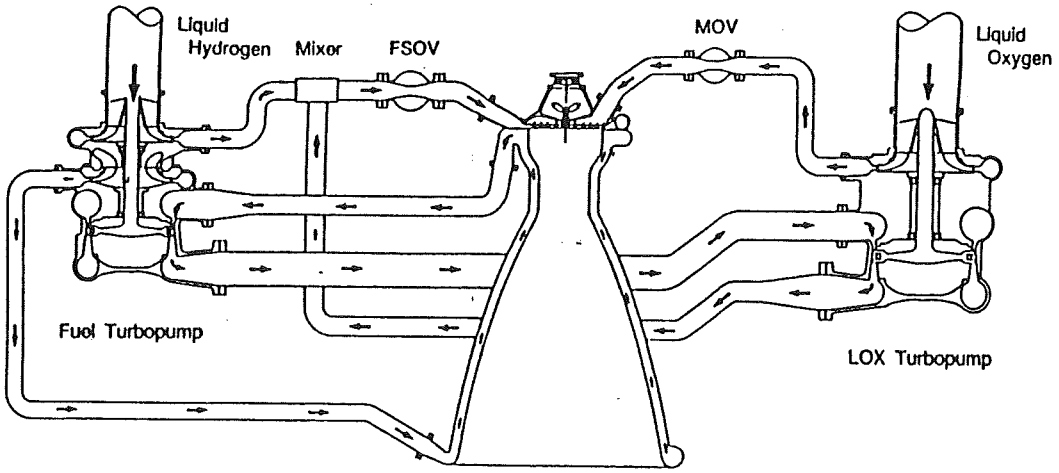
In the DUAL EXPANDER, both propellants are used to cool the combustion chamber and drive separate turbopump turbines - - - gaseous hydrogen to drive the fuel turbine and gaseous oxygen to drive the oxidizer turbine.

DUAL EXPANDER



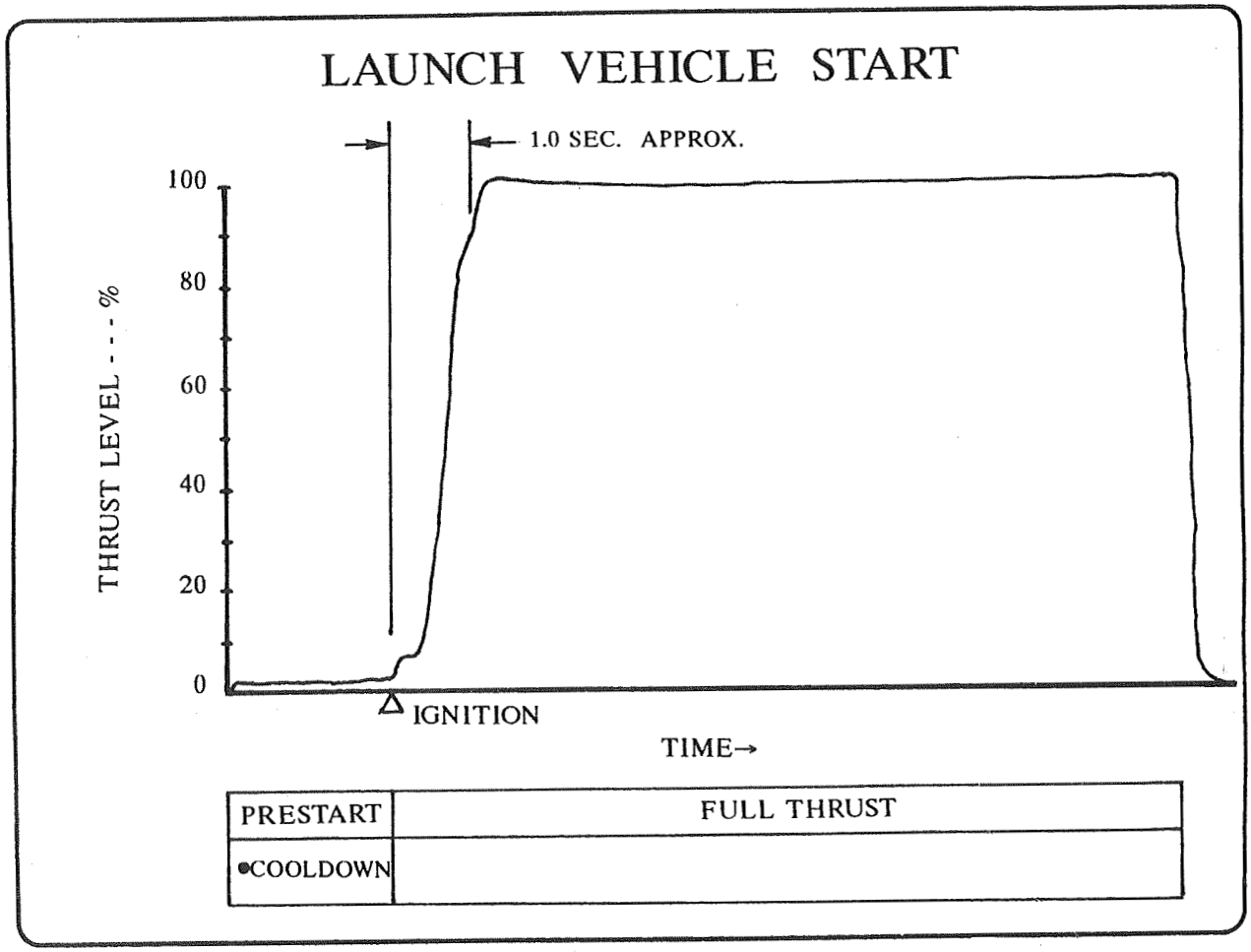
The SPLIT EXPANDER is a variation of the single expander. Like the single expander, hydrogen is used for coolant and the turbine drive gas; but unlike the single expander, not all the hydrogen flow is used. A portion of the hydrogen flow after the first-stage of the fuel pump is diverted directly to the combustion chamber. The remainder of the hydrogen flow is directed to subsequent stages of the fuel pump, through the chamber cooling passages, through the turbine(s), and then into the combustion chamber. The advantage of this arrangement is the reduced power requirement for the fuel pump and a resultant higher chamber pressure.

SPLIT EXPANDER



For launch vehicle applications where gravitational losses are of concern, it is desireable to ignite and accelerate the engine to rated thrust rapidly.

A cryogenic engine requires a short period (prestart) to cooldown the turbopumps prior to accelerating the engine. During prestart, the propellants flow through the engine turbopumps and are dumped overboard unburnt.

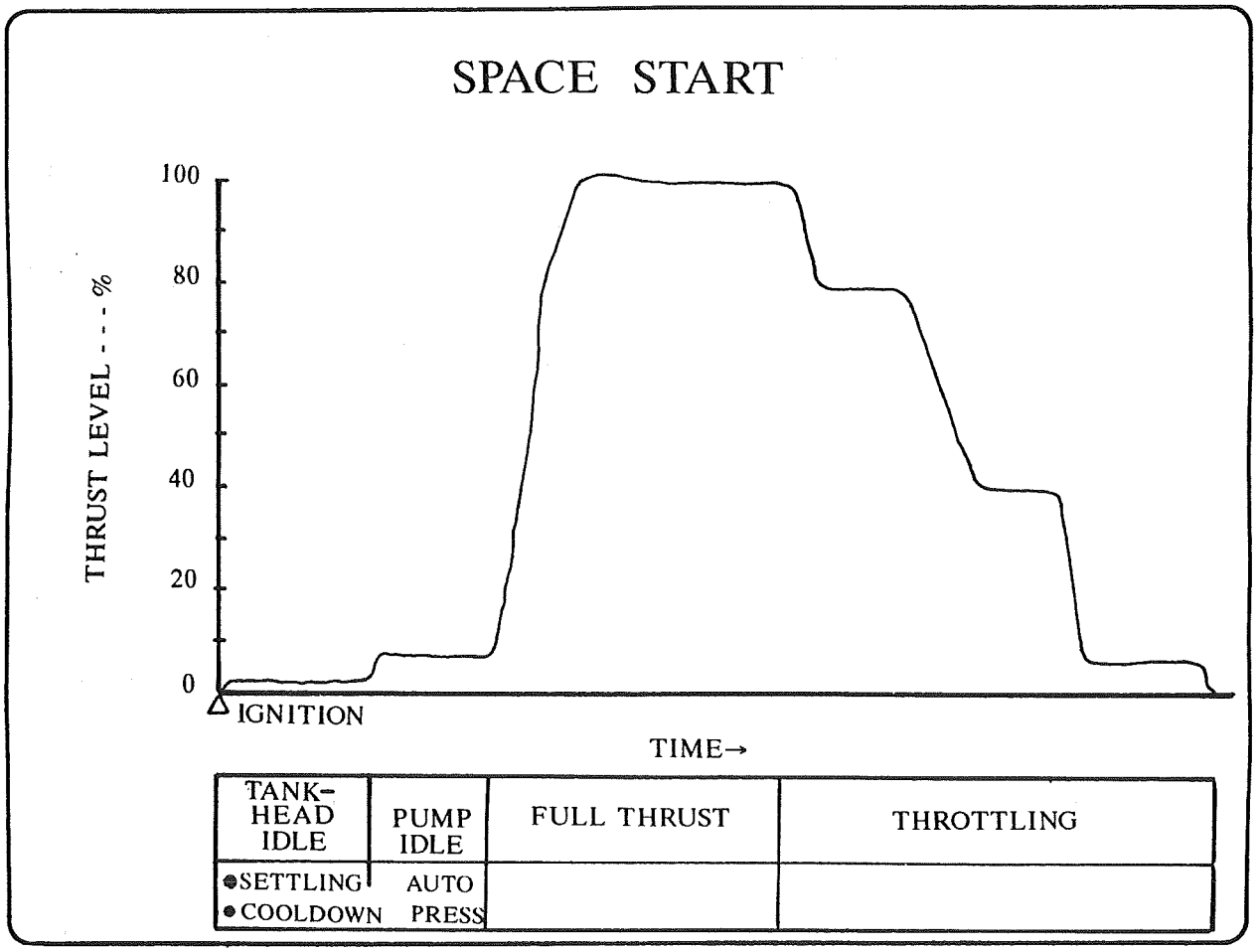


For a space application where gravitational losses are not as critical, a more efficient engine start sequence is possible.

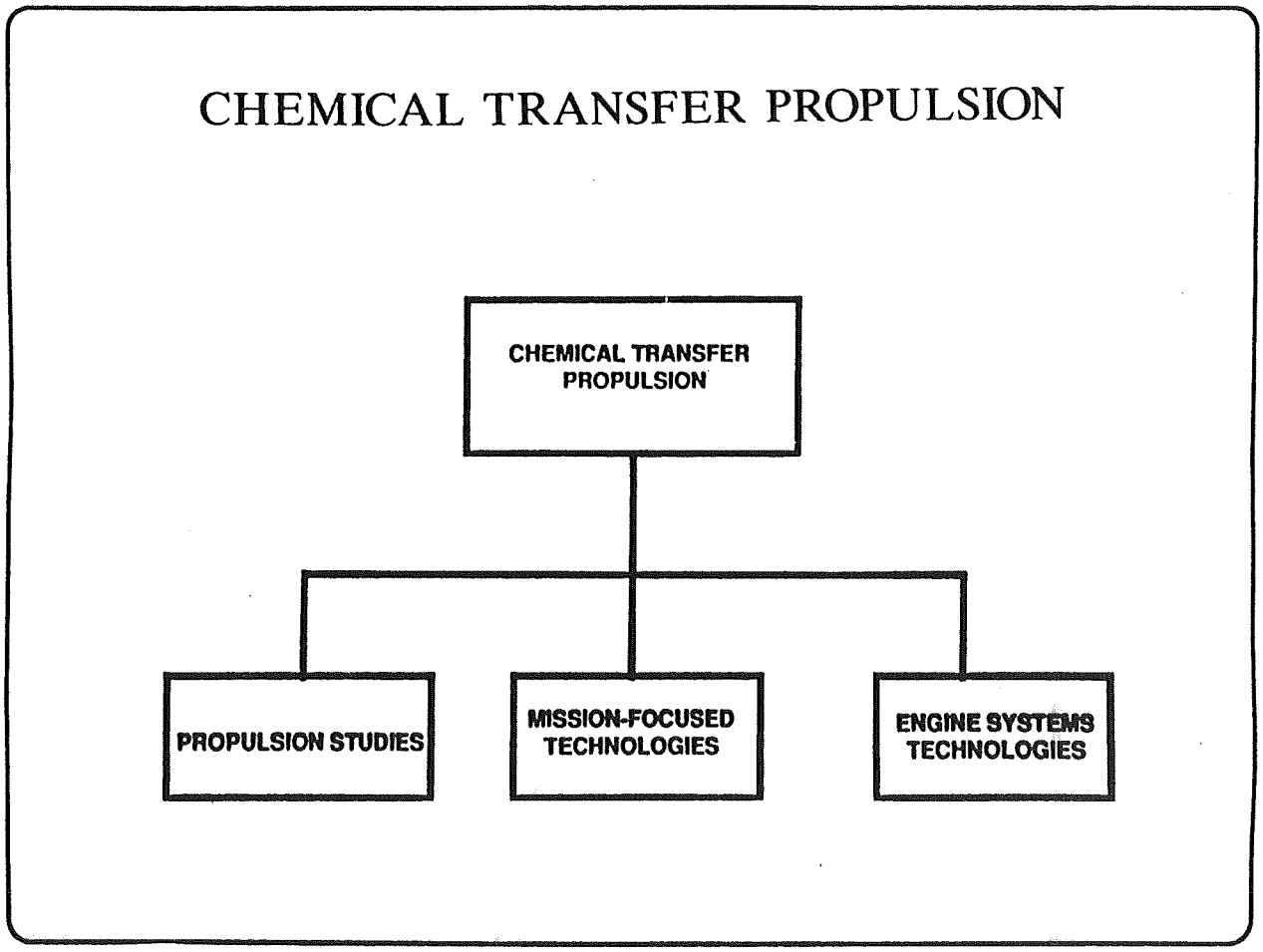
The engine would initially operate at TANKHEAD IDLE - - - engine pumps not rotating and the engine combustion chamber being supplied with vehicle tank pressure propellants (gaseous, liquid, or mixed-phase). Tankhead idle would be used to settle the propellants in the vehicle tanks and to cooldown the engine turbopumps. By burning the cooldown flow, significant total impulse could be realized.

After turbopump cooldown, the engine would then operate at PUMPED IDLE. During pumped idle, the turbopumps operate with zero NPSH propellants at the pump inlets. High pressure gases (GOX and GH2) are tapped off the engine for autogenous pressurization of the vehicle propellant tanks in preparation for acceleration to rated thrust.

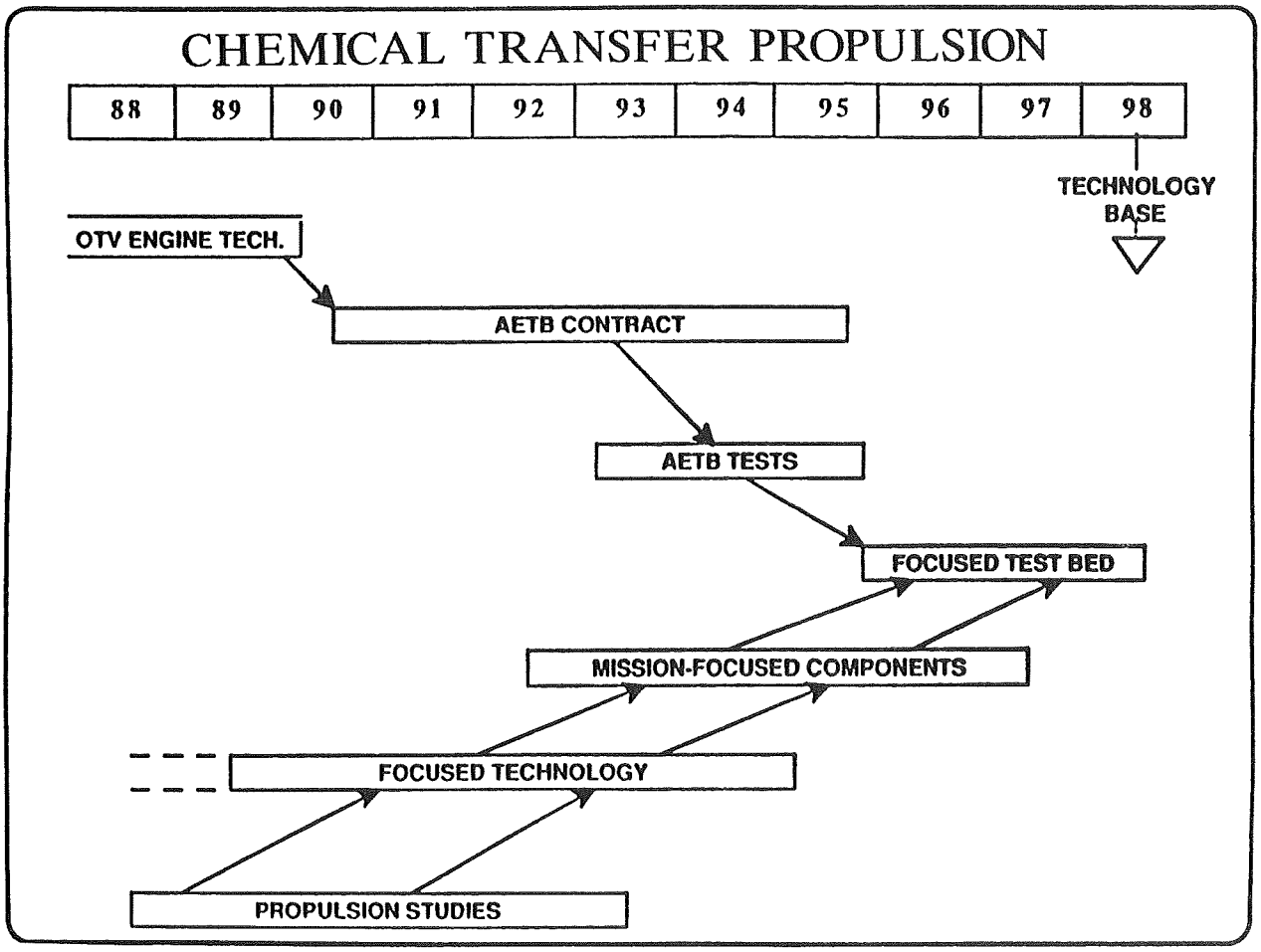
Besides the two idle modes, the engine is capable of deep throttling.



The objective of the current Chemical Transfer Propulsion (CTP) Program is the development of the technology base required to confidently initiate the actual ASE Development Program. The CTP Program has three work areas: Propulsion Studies, Mission-Focused Technologies, and Engine Systems Technologies.



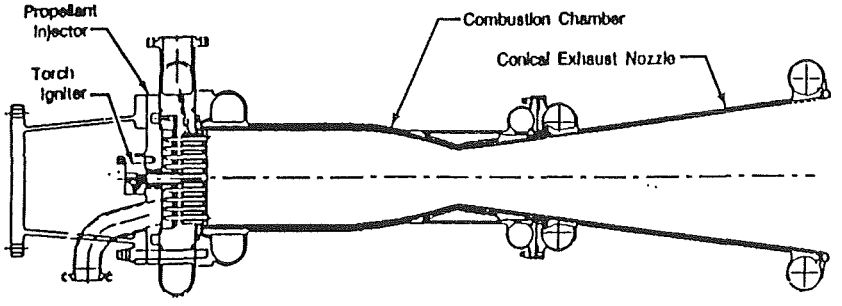
The three CTP work areas - - - Propulsion Studies, Mission-Focused Technologies (Focused Technology and Mission-Focused Components), and Engine Systems Technologies (AETB Contract and AETB Tests) - - - will lead to demonstration of the ASE technology base in a Focused Test Bed in the late 1990's.



A major portion of the CTP Program is the Engine Technologies and a major portion of the Engine Technologies is the Advanced Expander Test Bed (AETB) Engine.

Pratt & Whitney (West Palm Beach, FL) is under contract to NASA Lewis Research Center to design, build, test, and deliver two AETB's. The test bed engines will be tested at NASA to investigate system interactions and dynamics and to test Mission-Focused Components from other NASA contracts.

ADVANCED EXPANDER TEST BED (AETB)



Pc = 1200 psia

F = 16,000 lb

THROTTLING = 20:1

AREA RATIO = 7.5

LENGTH = 48 inches

O/F = 5 to 12

- SPLIT EXPANDER CYCLE
- TANKHEAD AND PUMPED IDLE MODES
- 3 STAGE LH2 TURBOPUMP
 - COUNTER-ROTATING, BACK-BACK DUAL SPOOL
- 2 STAGE LO2 TURBOPUMP
- MILLED COPPER THRUST CHAMBER

ADVANCED MANNED LAUNCH SYSTEMS (AMLS)

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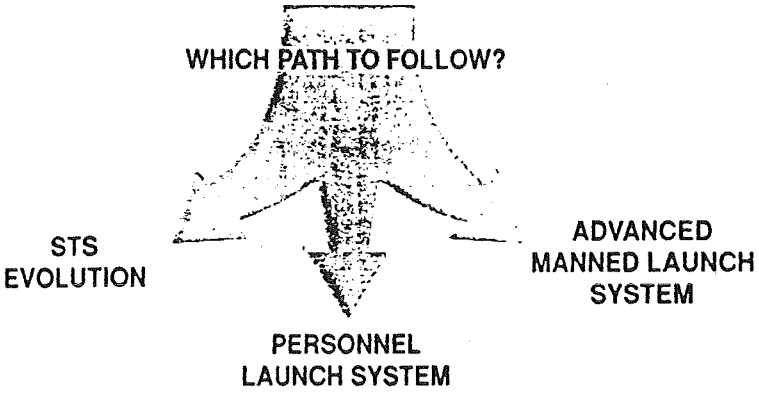
PROPULSION STUDIES FOR ADVANCED MANNED LAUNCH SYSTEMS

Vehicle Analysis Branch
Space Systems Division
NASA Langley Research Center

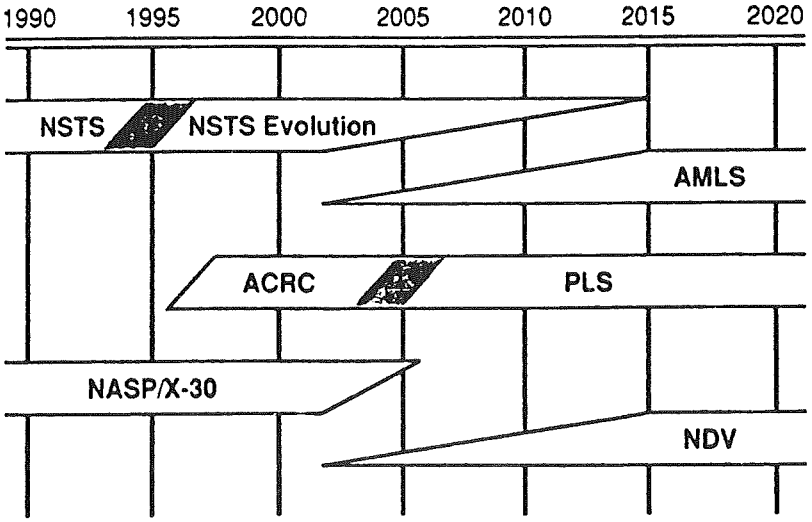
Presented by: D. Freeman

THE NEXT MANNED SPACE TRANSPORTATION SYSTEM

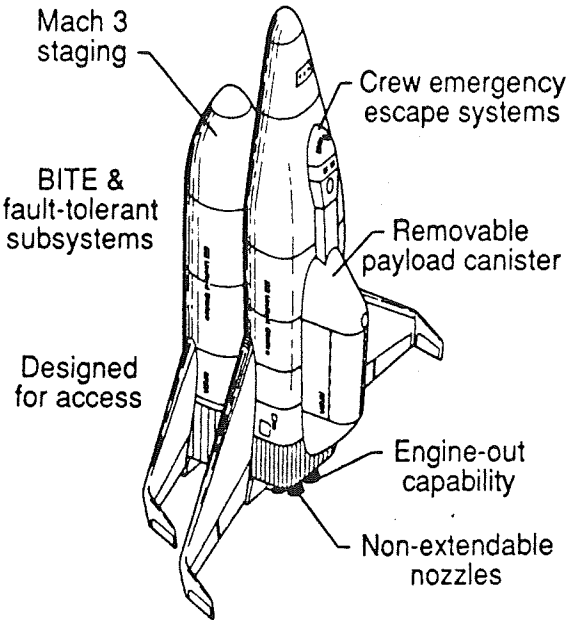
- Satisfy people/payload requirements
- Improve cost effectiveness
- Increase reliability
- Increase margins



MANNED SPACE TRANSPORTATION OPTIONS SCHEDULE



DESIGN FOR OPERATIONS, RELIABILITY AND SAFETY

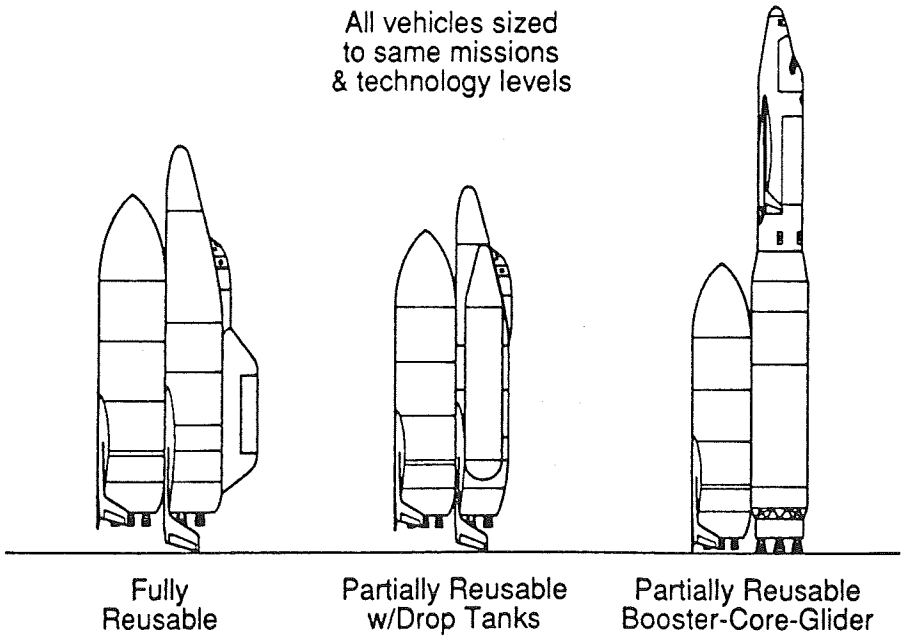


Technology Advantage Applied to:

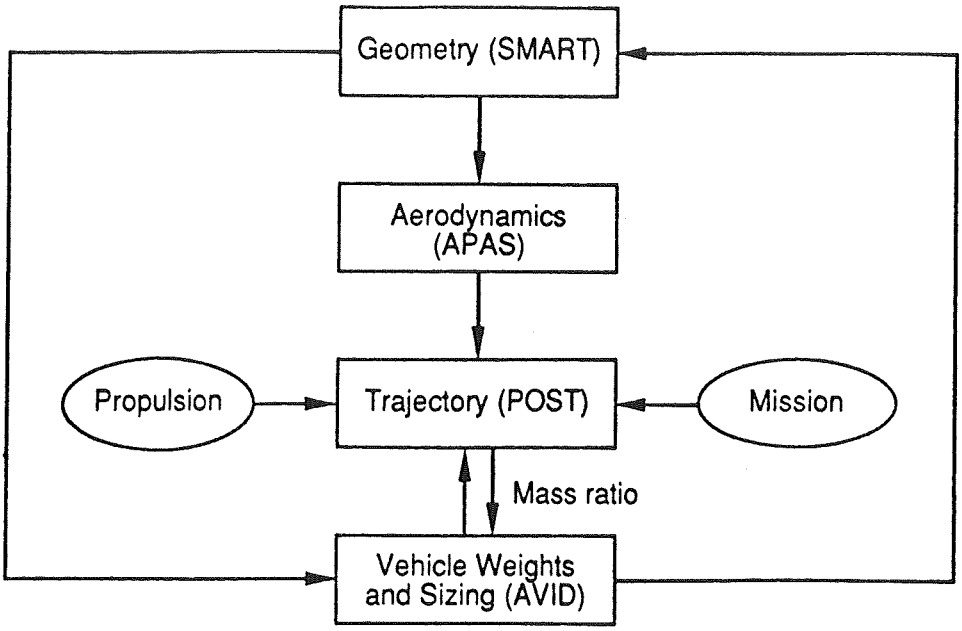
- Operations streamlining
- Robust subsystems
- Improved reliability
- Assured mission success
- Safety

Not Maximum Payload

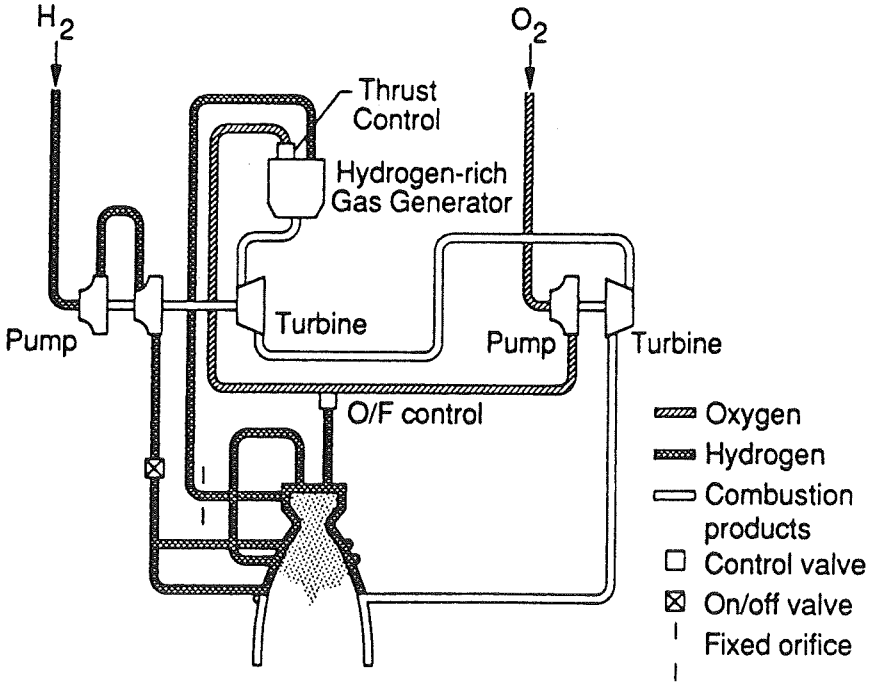
ADVANCED MANNED LAUNCH SYSTEM CONCEPTS



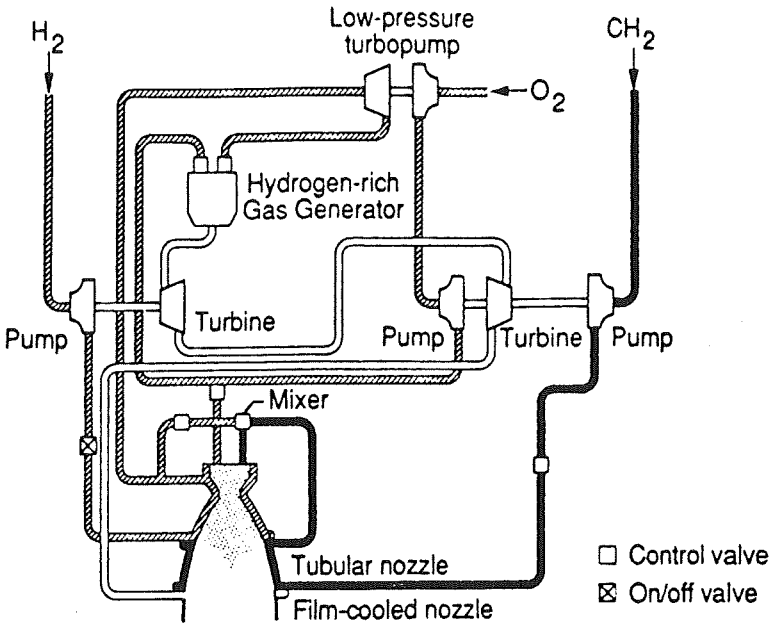
VEHICLE DESIGN PROCESS



REFERENCE STME ENGINE

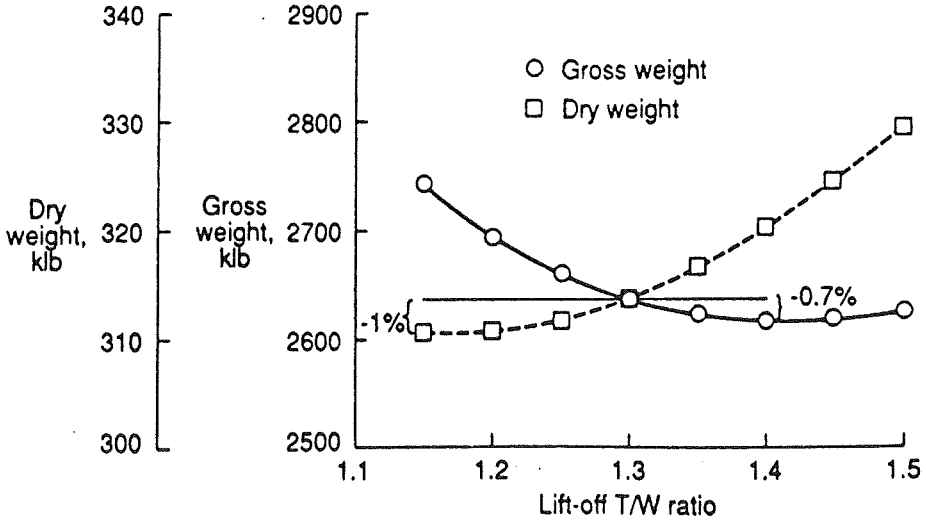


REFERENCE STBE ENGINE



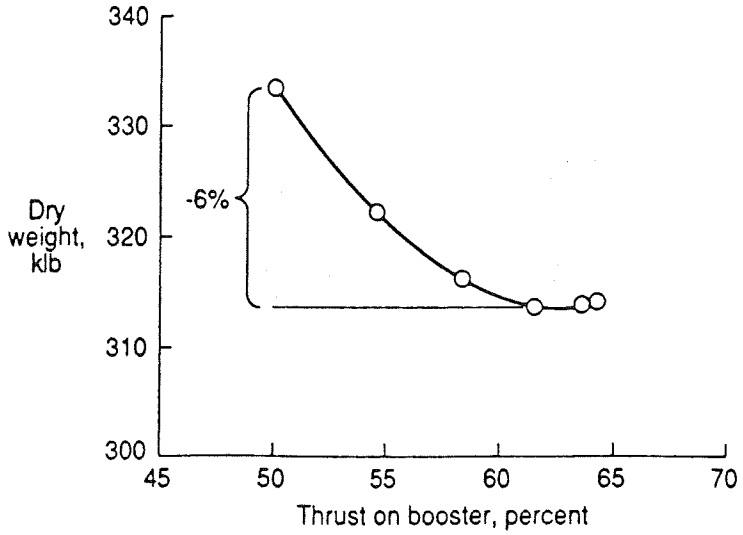
LIFT-OFF THRUST-TO-WEIGHT TRADE

FULLY REUSABLE, ALL LOX/LH_2



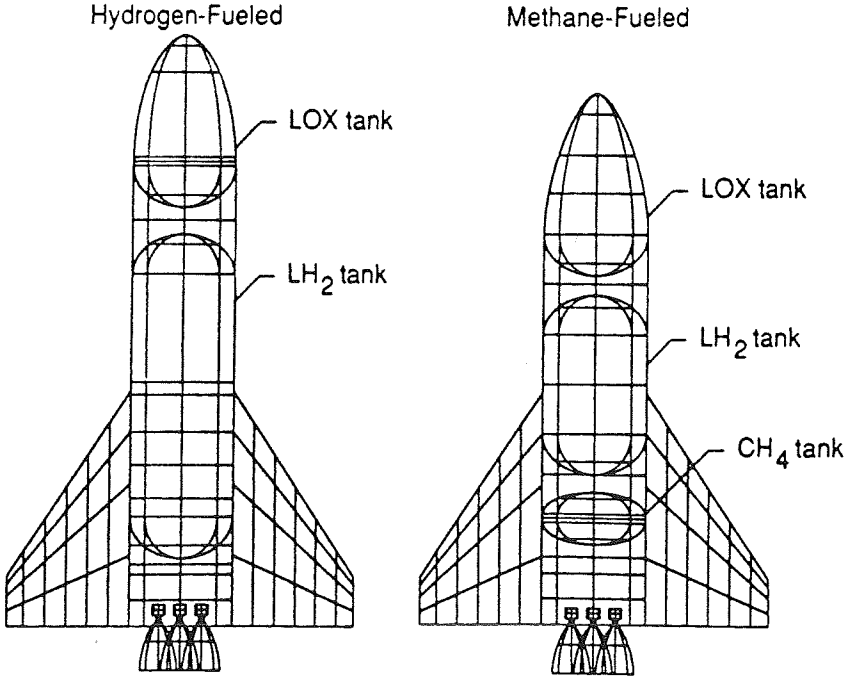
THRUST SPLIT TRADE AT LIFT-OFF

FULLY REUSABLE, ALL LOX/LH_2



- Chose 7 engines on booster, 4 engines on orbiter

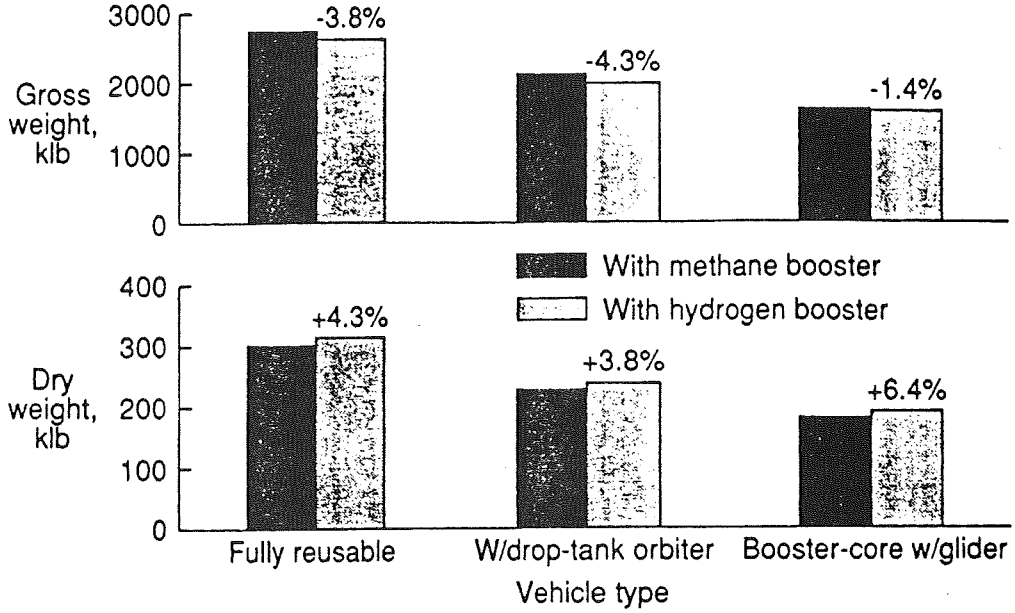
AMLS REUSABLE BOOSTERS



AMLS CONCEPT PROPULSION TRADES

SINGLE FUEL VERSUS DUAL FUEL

- All vehicles designed to same reference mission (polar, 12 klb) and same technology level
- Boosters use methane or hydrogen as main propellant (STME/STBE engine)

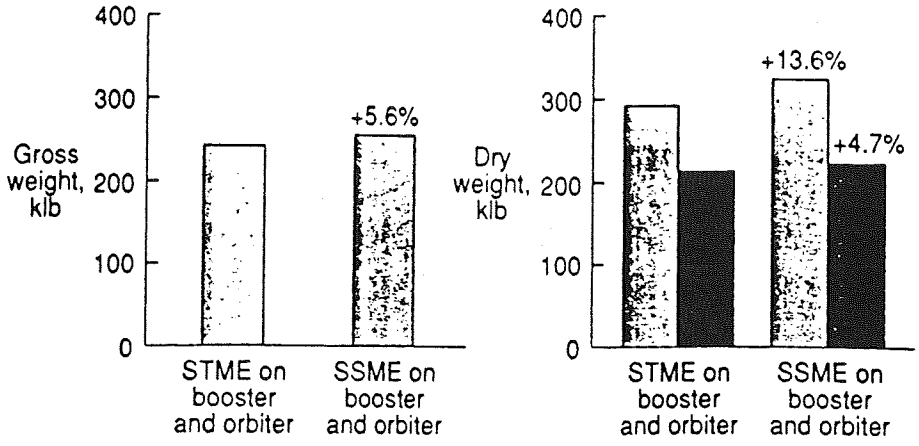


ADVANTAGES OF THE ALL-HYDROGEN VEHICLE

- Reduced development costs
 - Delete STBE-type engine development (traded off against slightly increased vehicle dry weight)
- Reduced production costs
 - Increased line production of one type of engine
- Simpler operations
 - Common engine systems used on both stages
 - Elimination of hydrocarbon fuel and associated storage, handling, and management organization structure
- Environmental factors
 - Hydrogen fuel cleaner burning
 - Reduced engine maintenance
 - Elimination of detrimental hydrocarbon exhaust byproducts

SSME VERSUS SINGLE-POSITION STME

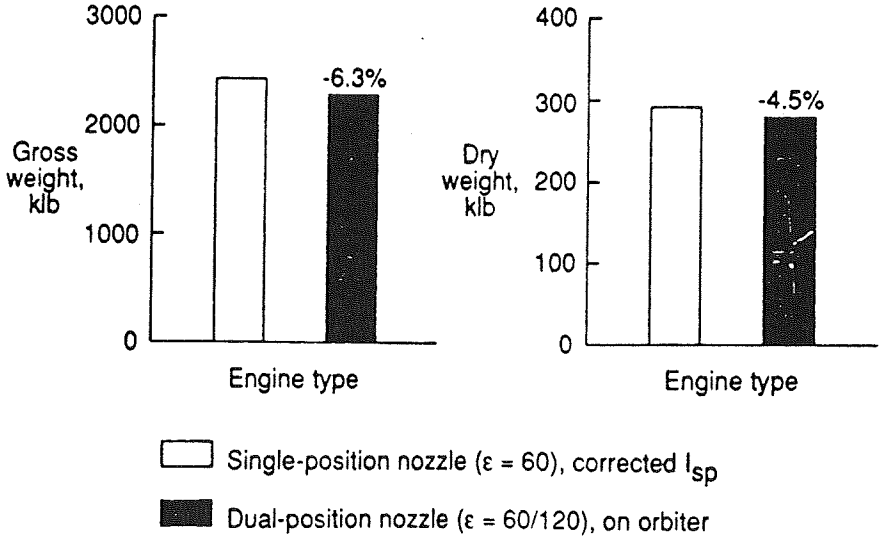
FULLY REUSABLE, ALL LOX/LH_2 VEHICLE



- Cases use current, unmodified SSME
- OF ratio is 6.0 for SSME and STME
- ε = 77.5 for SSME, ε = 60 for STME
- Both cases have engine-out capability

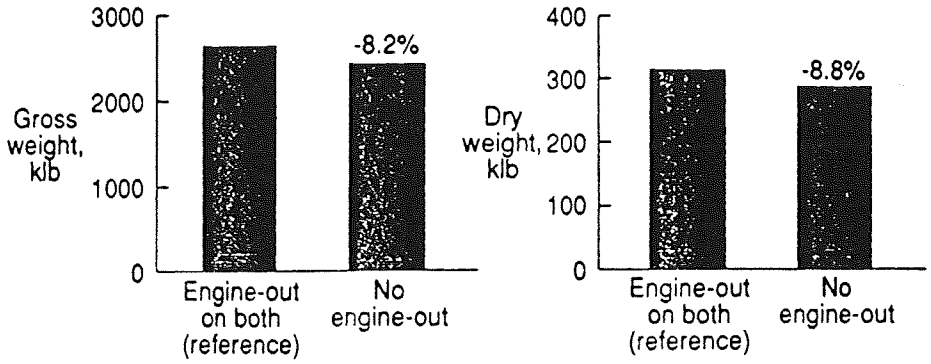
DUAL-POSITION NOZZLE TRADE

FULLY REUSABLE, ALL LOX/LH_2



ENGINE-OUT CAPABILITY TRADE

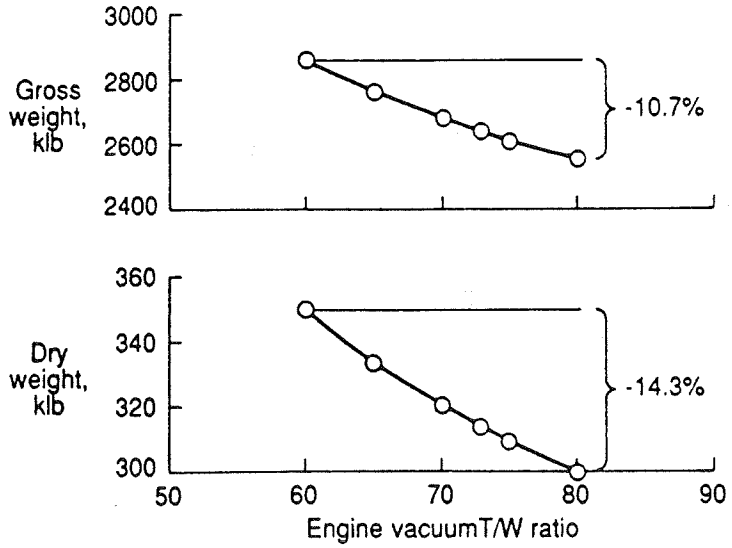
FULLY REUSABLE, ALL LOX/LH_2 VEHICLE



- At least 4 engines required on both the booster and orbiter
- Increased vehicle reliability brings about:
 - Quantitative reduction in recurring costs
 - Qualitative increase in crew and mission safety

ENGINE THRUST-TO-WEIGHT RATIO TRADE

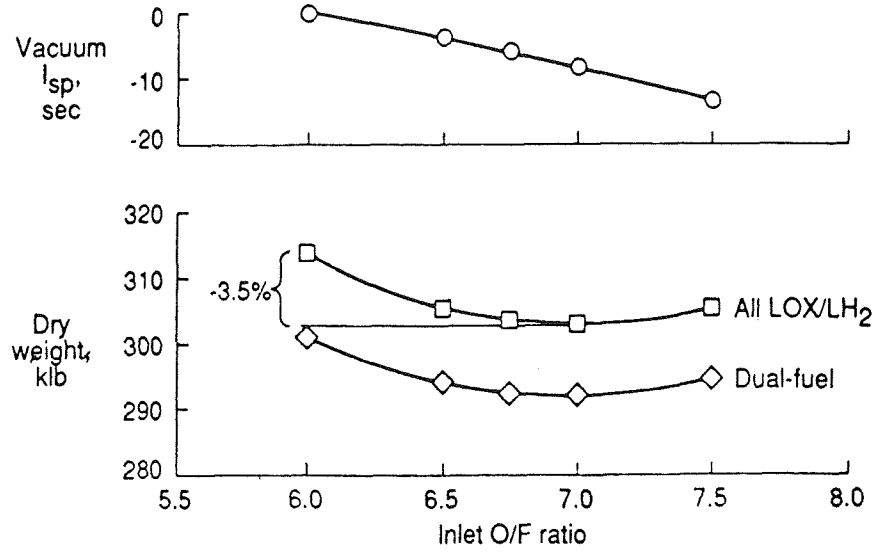
FULLY REUSABLE, ALL LOX/LH_2



- Constant O/F ratio and I_{sp} for all cases

AMLS OXIDIZER/FUEL RATIO TRADE

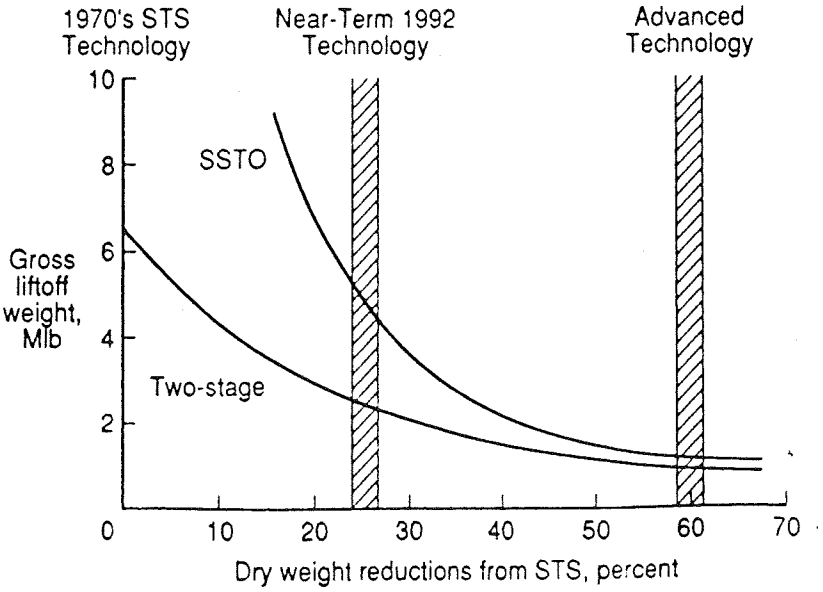
FULLY REUSABLE



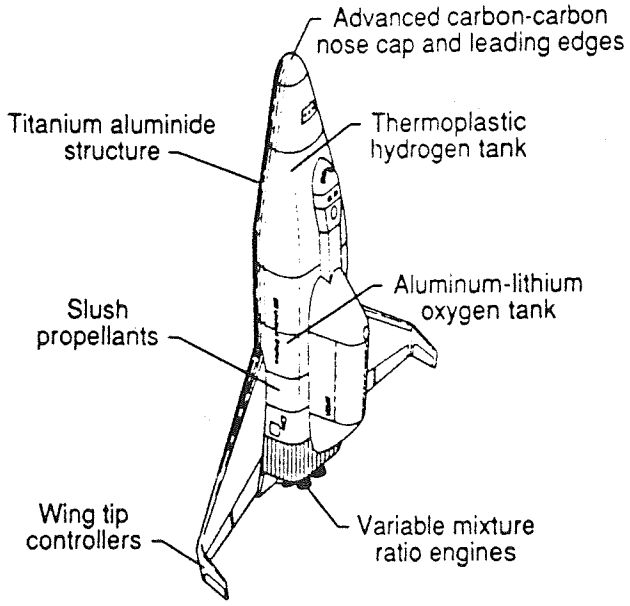
CONCLUSIONS

- Development of a new hydrocarbon booster engine (like the STBE) for next-generation manned systems may not be cost effective
- Development of a new hydrogen engine (like the STME) for next-generation manned systems could prove cost effective for use as a main (and booster) propulsion system
- Use of a dual-position nozzle would probably not be beneficial for a design-for-operations system like AMLS
- An increase in oxidizer-to-fuel ratio from the current SSME level of 6 to approximately 7 would be beneficial in reducing future launch system weights

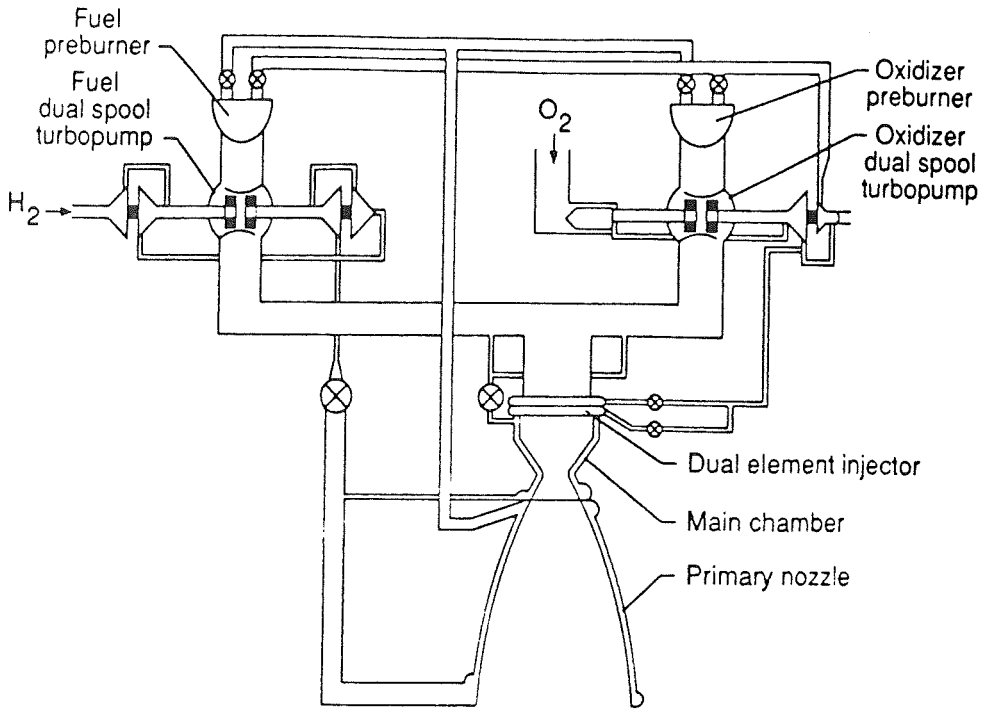
TECHNOLOGY EFFECT ON ROCKET LAUNCH VEHICLE WEIGHT



ADVANCED SSTO VEHICLE TECHNOLOGIES



PRATT & WHITNEY VMR FLOW SCHEMATIC



ADVANCED VARIABLE-MIXTURE RATIO ENGINE

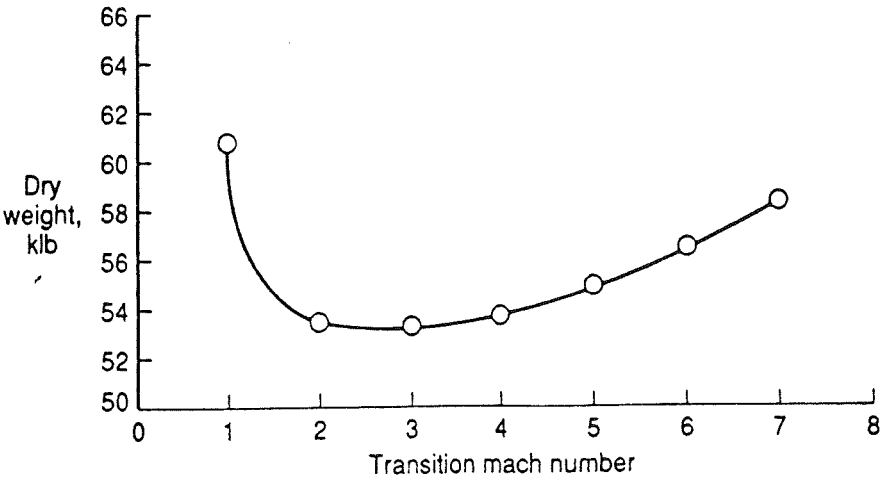
HYDROGEN/OXYGEN

Mode	1	2	SSME (109%)
O/F Ratio	12	6	6.026
Nozzle	Retracted	Extended	Single-position
Expansion Ratio	40	150	77.5
Vacuum Thrust, lb	254,500	176,900	512,300
Vacuum Isp, sec	362	467	452
Chamber Press., psia	4,000	2,700	3260
SL Thrust, lb	234,580	142,832*	417,300
SL Thrust/Weight	109.5	66.68*	60

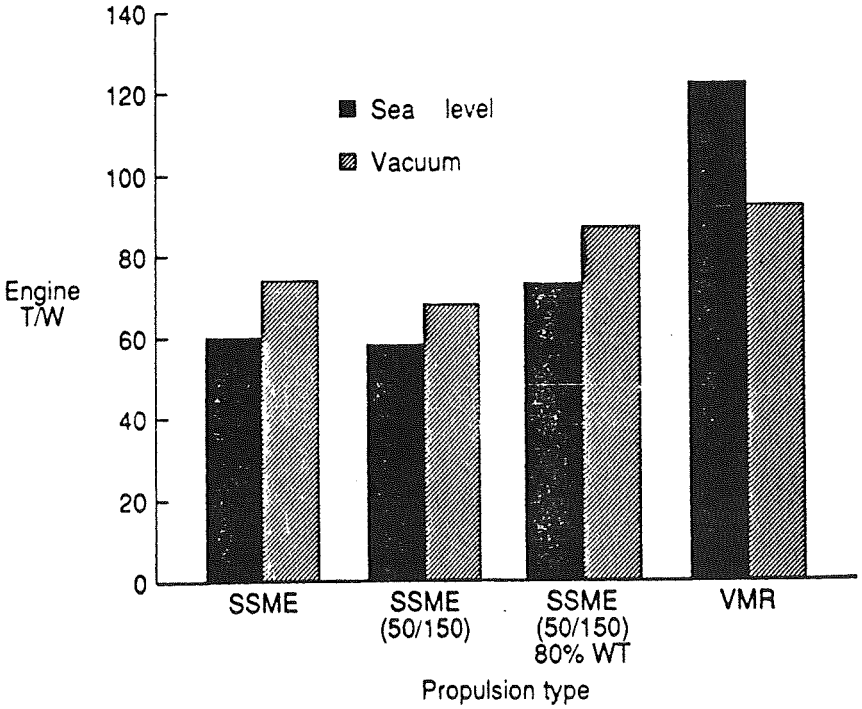
*Area ratio of $\varepsilon = 40$

TRANSITION MACH NUMBER TRADE (VMR ENGINE)

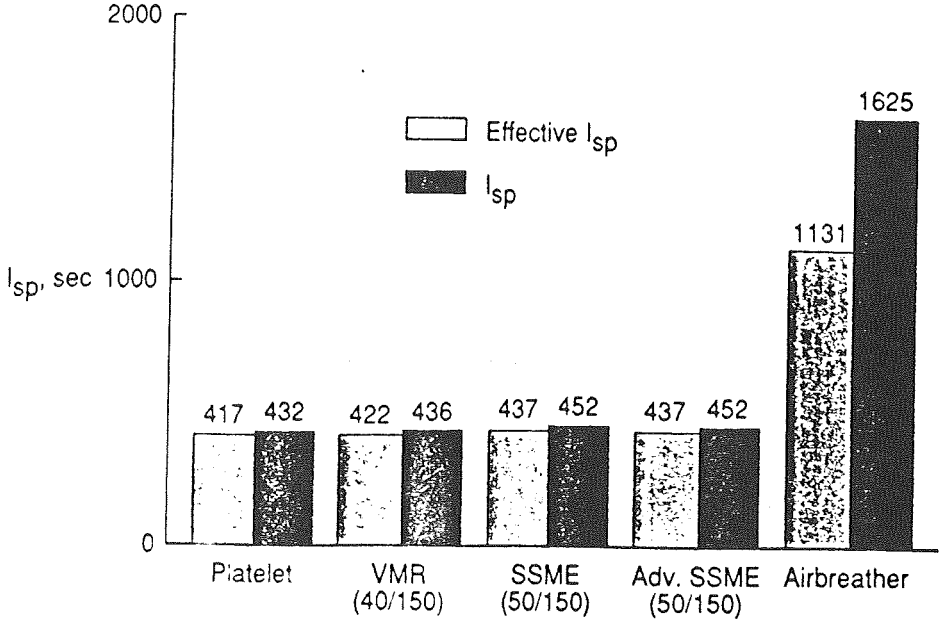
INITIAL TRADE



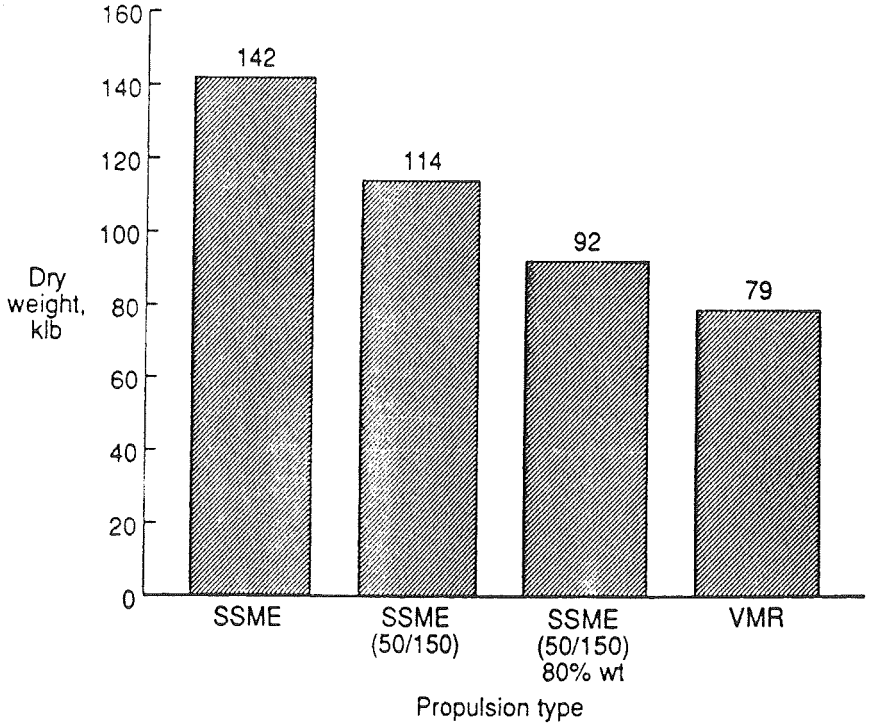
COMPARISON OF PROPULSION CHARACTERISTICS



TIME AVERAGED SPECIFIC IMPULSE



DRY WEIGHT SENSITIVITY TO PROPULSION TYPE



CONCLUSIONS

- Application of advanced technologies could allow introduction of rocket-powered SSTO vehicle for 2015 IOC
 - Low dry weight compared to two-stage and airbreathers
 - Lower operation costs than two-stage

- Application of variable-mixture-ratio technology and cooled, vaneless turbines could greatly benefit advanced vehicles
 - Lower specific impulse
 - Higher T/W ratio
 - Higher bulk density

PRESENTATION 1.3.9

NATIONAL AEROSPACE PLANE (NASP)

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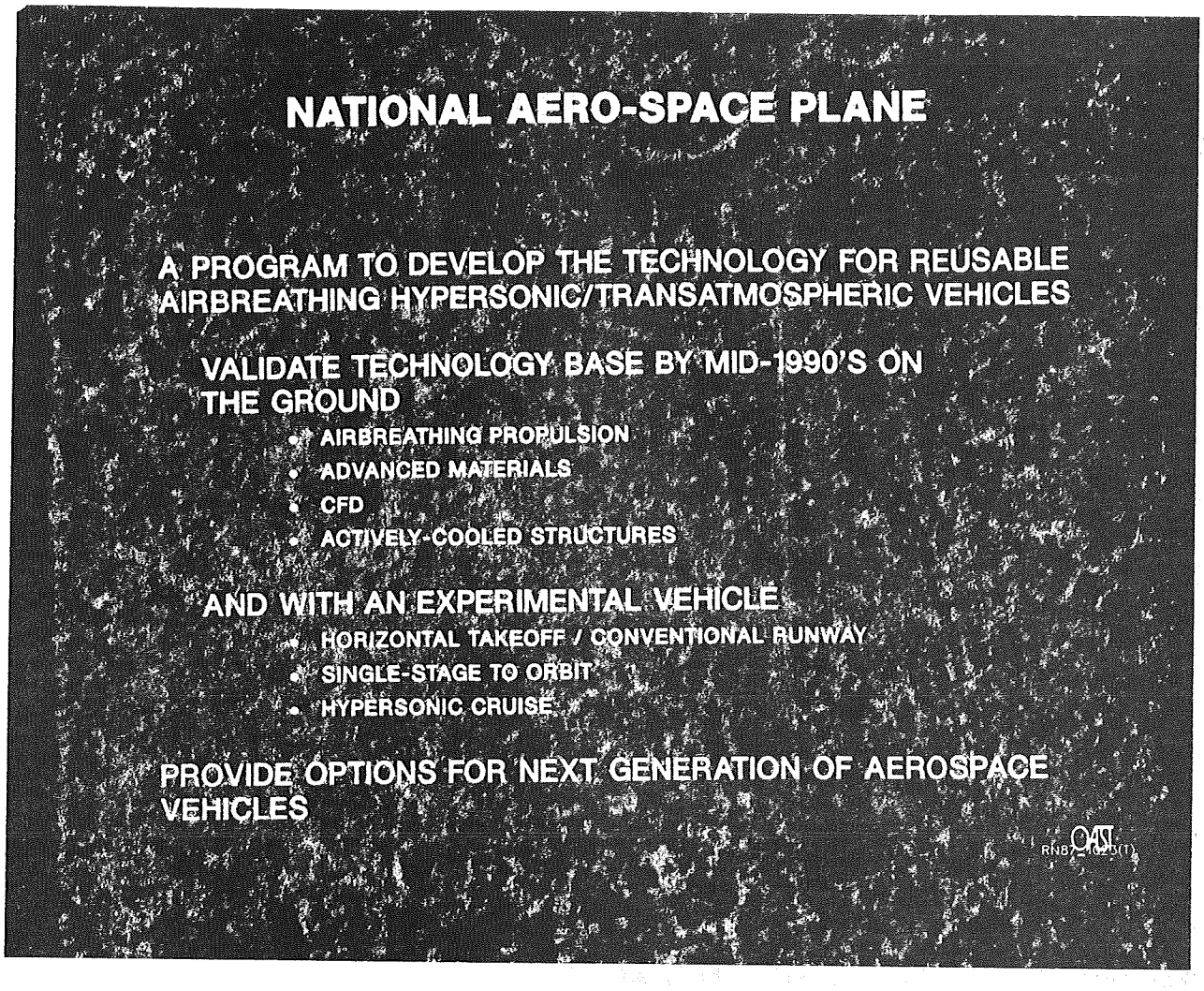
900
540718
N91-28214

NATIONAL AERO-SPACE PLANE (NASP) PROGRAM

PENNSYLVANIA STATE UNIVERSITY
JUNE 26-29, 1990

MING H. TANG
Deputy Director, NASP NIO
Pentagon

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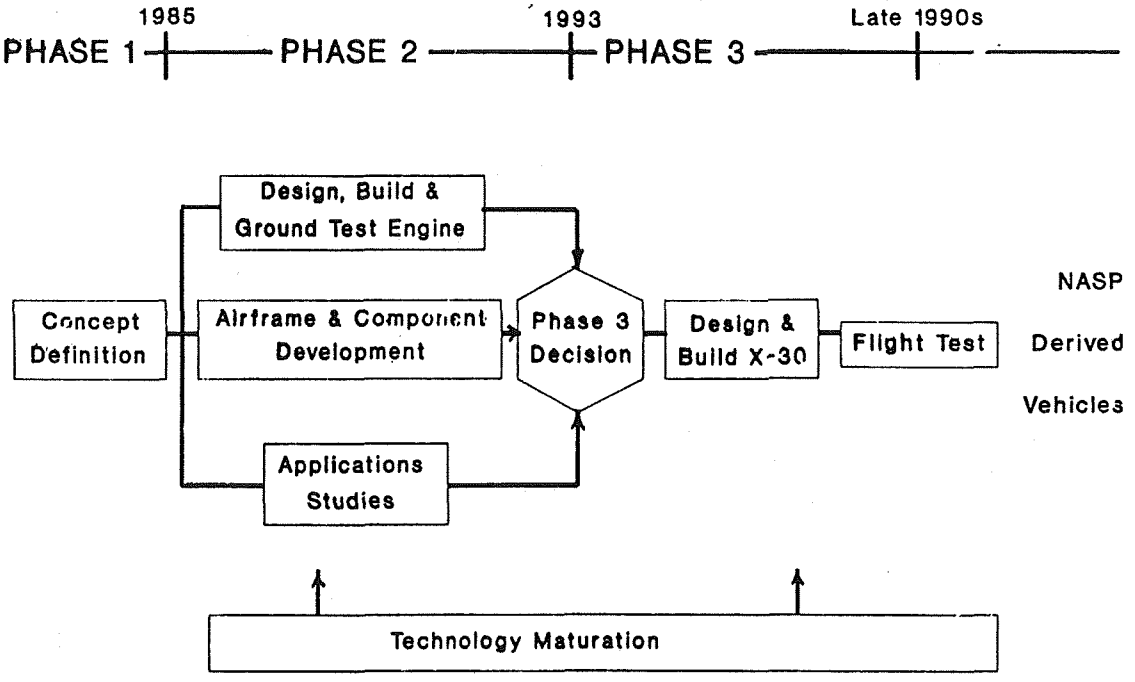


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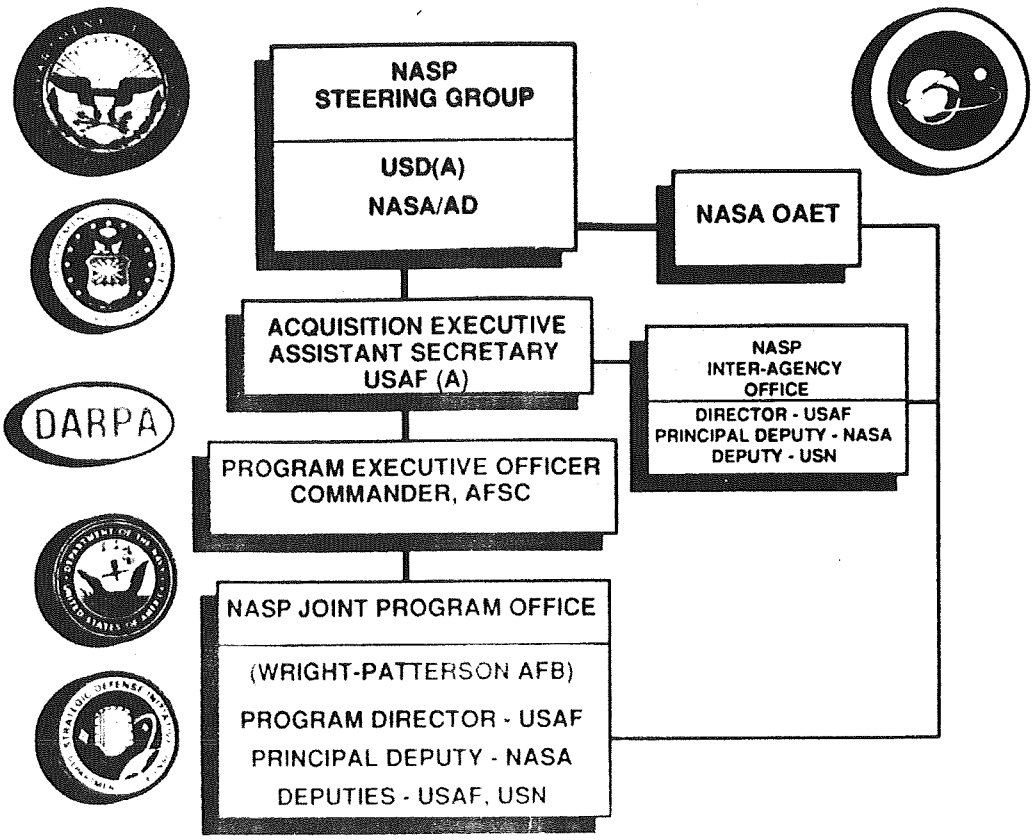
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NASP PROGRAM SCHEDULE

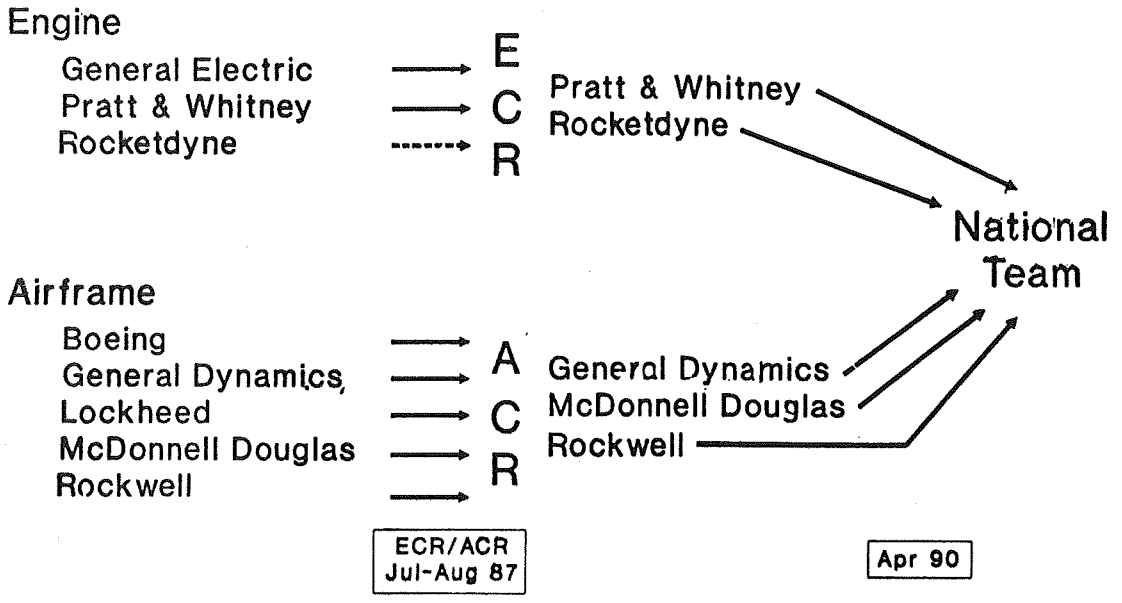


NATIONAL AERO-SPACE PLANE PROGRAM ORGANIZATION



Competitive Strategy

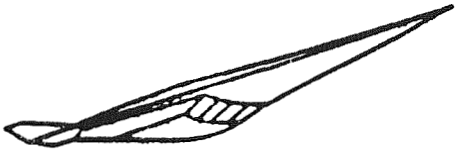
Phase 2



ALTITUDE (1000 FT)
350
300
250
200
150
100
50
0
0
5
10
15
20
25
30
MACH NUMBER
SHUTTLE ASCENT
SHUTTLE ENTRY
NASP FLIGHT ENVELOPE
AIRBREATHING FLIGHT CORRIDOR

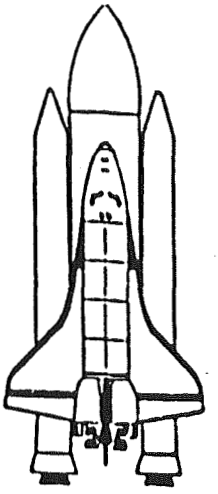
AEROSPACE PLANE
SHUTTLE COMPARISONS

AEROSPACE PLANE



- SINGLE STAGE TO ORBIT
- AIR-BREATHING PROPULSION
- HORIZONTAL TAKE-OFF FROM CONVENTIONAL RUNWAY
- ORBIT ON DEMAND
- ALTERNATE MISSION: HYPERSONIC CRUISE

SPACE SHUTTLE



- MULTI-STAGE VEHICLE
- ROCKET PROPULSION
- VERTICAL TAKE-OFF - SPECIALIZED LAUNCH REQMTS
- WEEKS FOR LAUNCH PREPARATION
- NO ALTERNATE MISSION CAPABILITY

NASP CONFIGURATION MATRIX

WING / BODY



Advantages
- -Low speed aero
- -Tankage design

Disadvantages
- -Wing/inlet coupling
- -Overexpanded flow to inlet

BLENDED BODY



Advantages
- -Precompression efficiency
- -Structural weight

Disadvantages
- -Low speed aero
- -Elliptical tanks

CONICAL ACCELERATOR



Advantages
- -Thrust margin
- -Precompression efficiency
- -Tankage design

Disadvantages
- -Sensitivity to angle of attack
- -Cruise efficiency

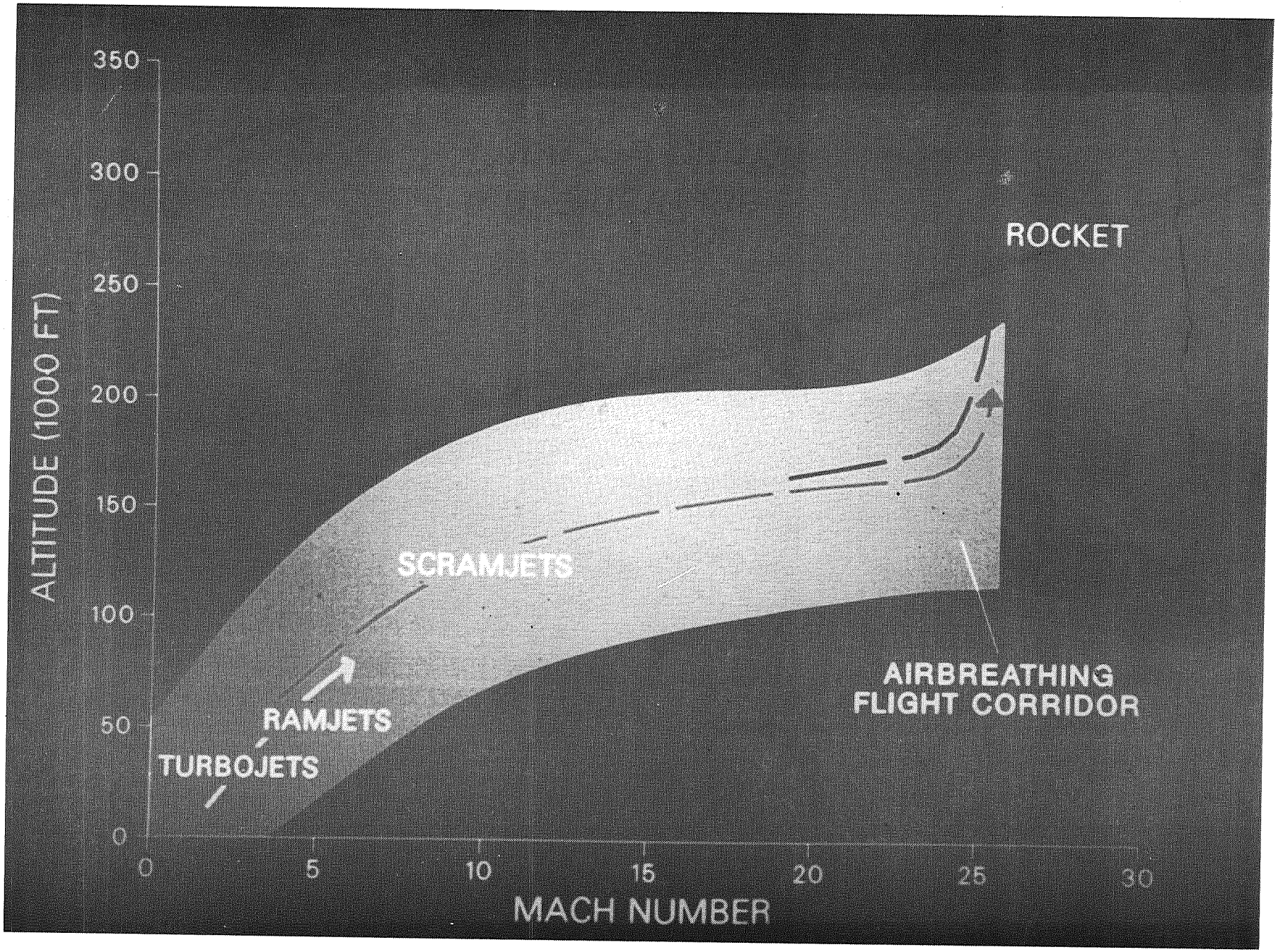
CONFINED FLOW FIELD



Advantages
- -Precompression efficiency
- -Aero efficiency

Disadvantages
- -Structural weight
- -Thermal protection
- -Off-design sensitivity

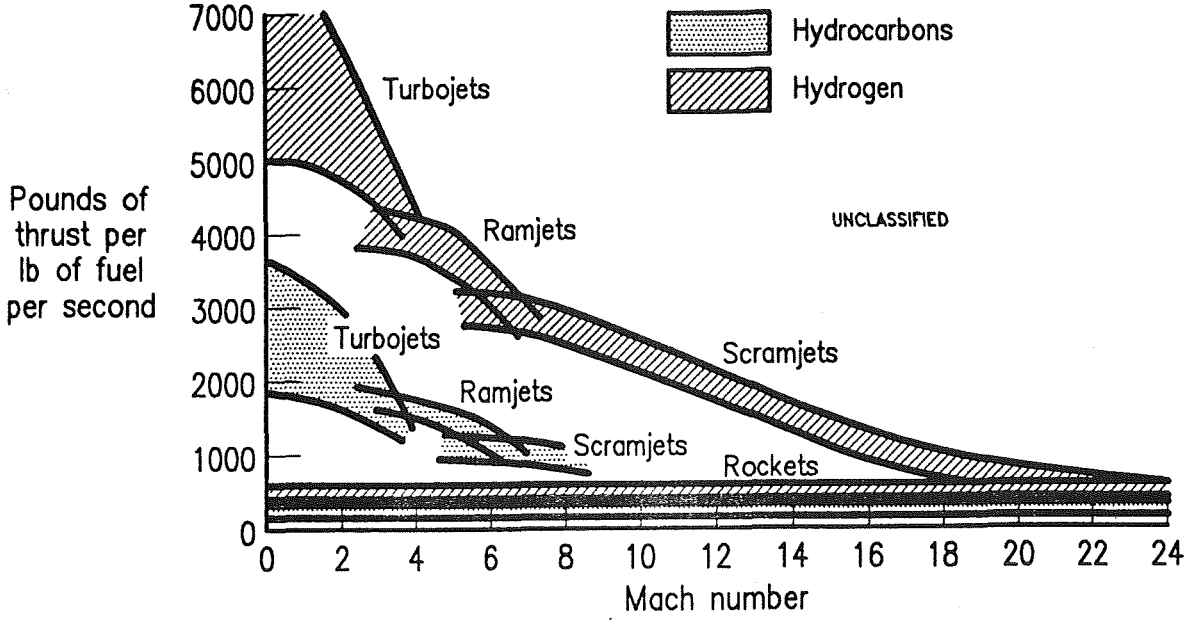
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ALTITUDE (1000 FT)
350
300
250
200
150
100
50
0
0
5
10
15
20
25
30
MACH NUMBER
ROCKET
SCRAMJETS
RAMJETS
TURBOJETS
AIRBREATHING
FLIGHT CORRIDOR

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PROPULSION OPTIONS (U)

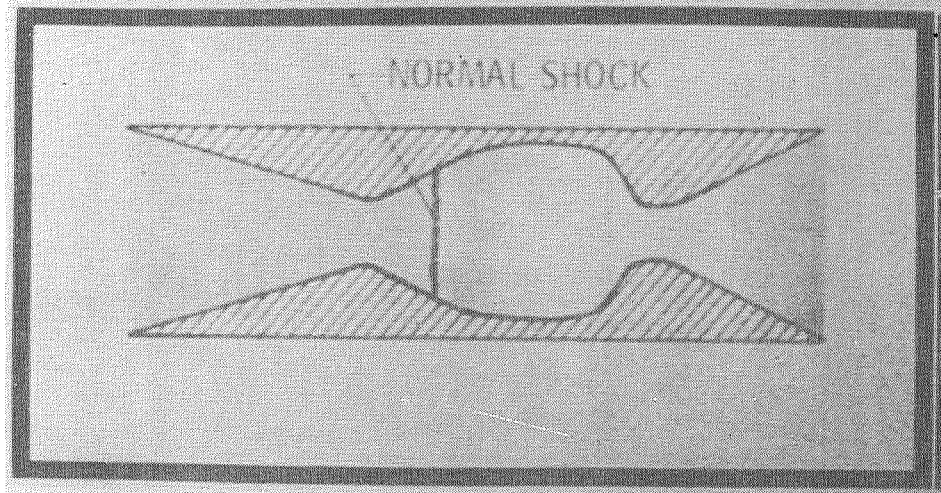
Fuel consumption comparison



UNCLASSIFIED

RAMJET

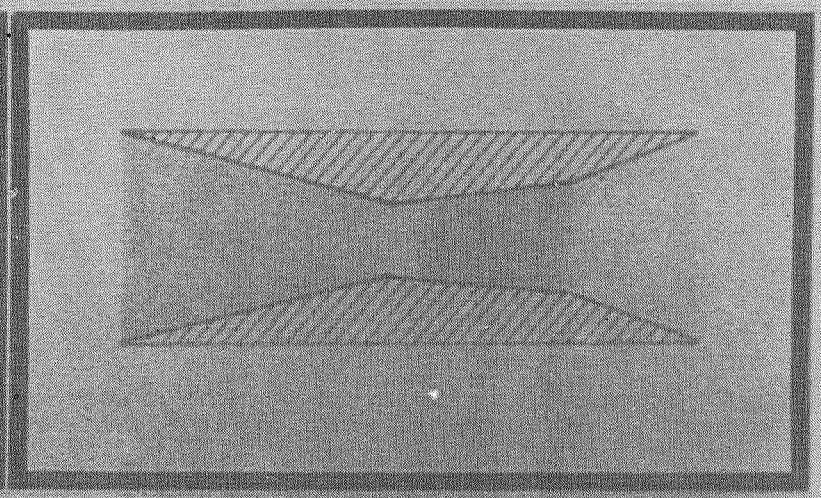
MACH 2 TO 4



SUBSONIC COMBUSTION

SCRAMJET

MACH 4 TO ?



SUPERSONIC COMBUSTION

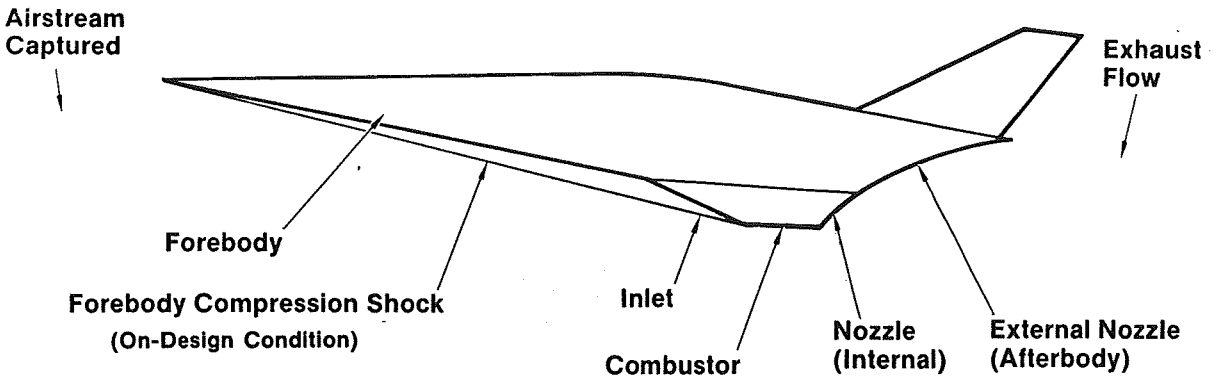
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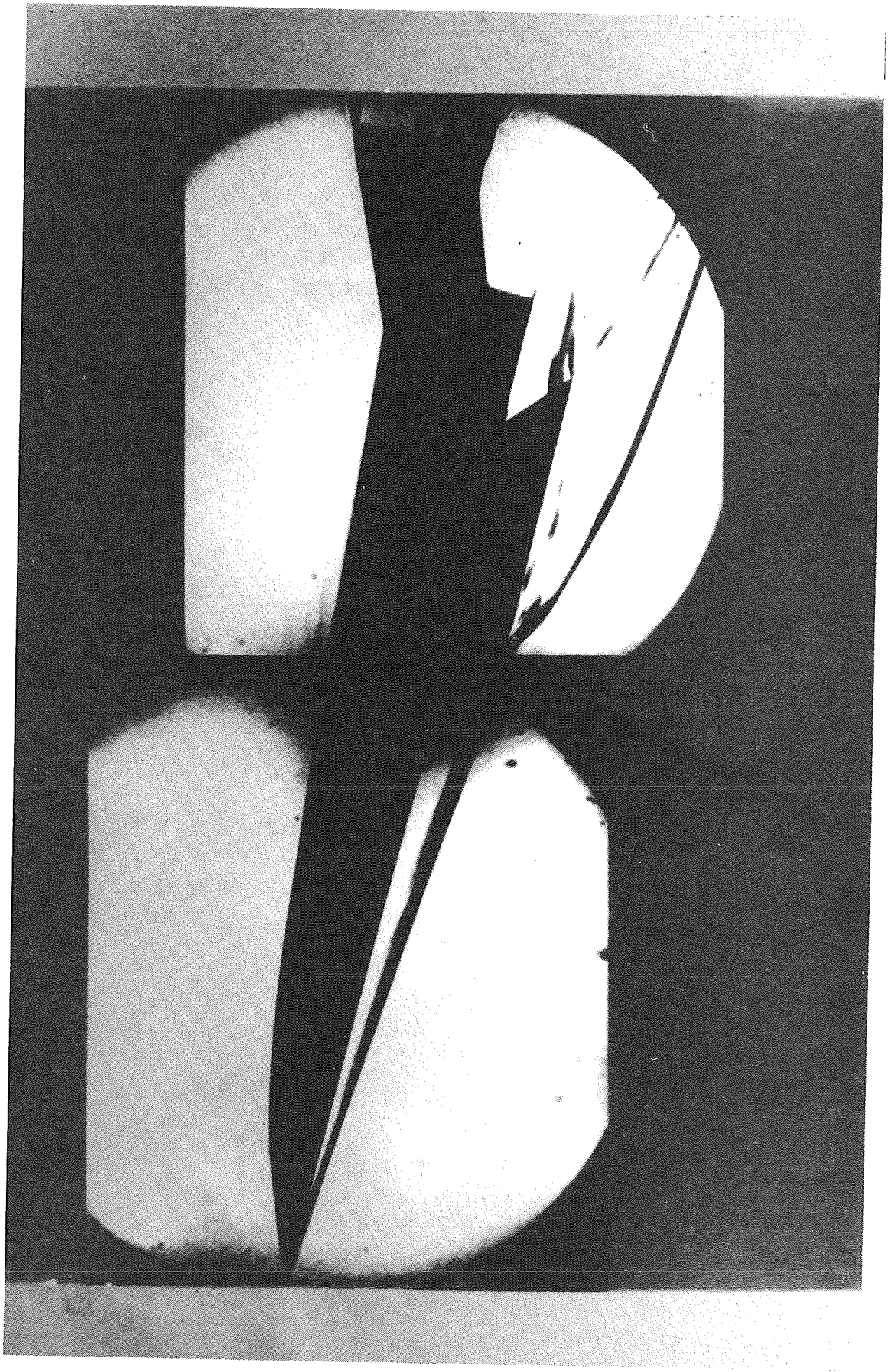
Propulsion Concept
High-Speed Scramjet

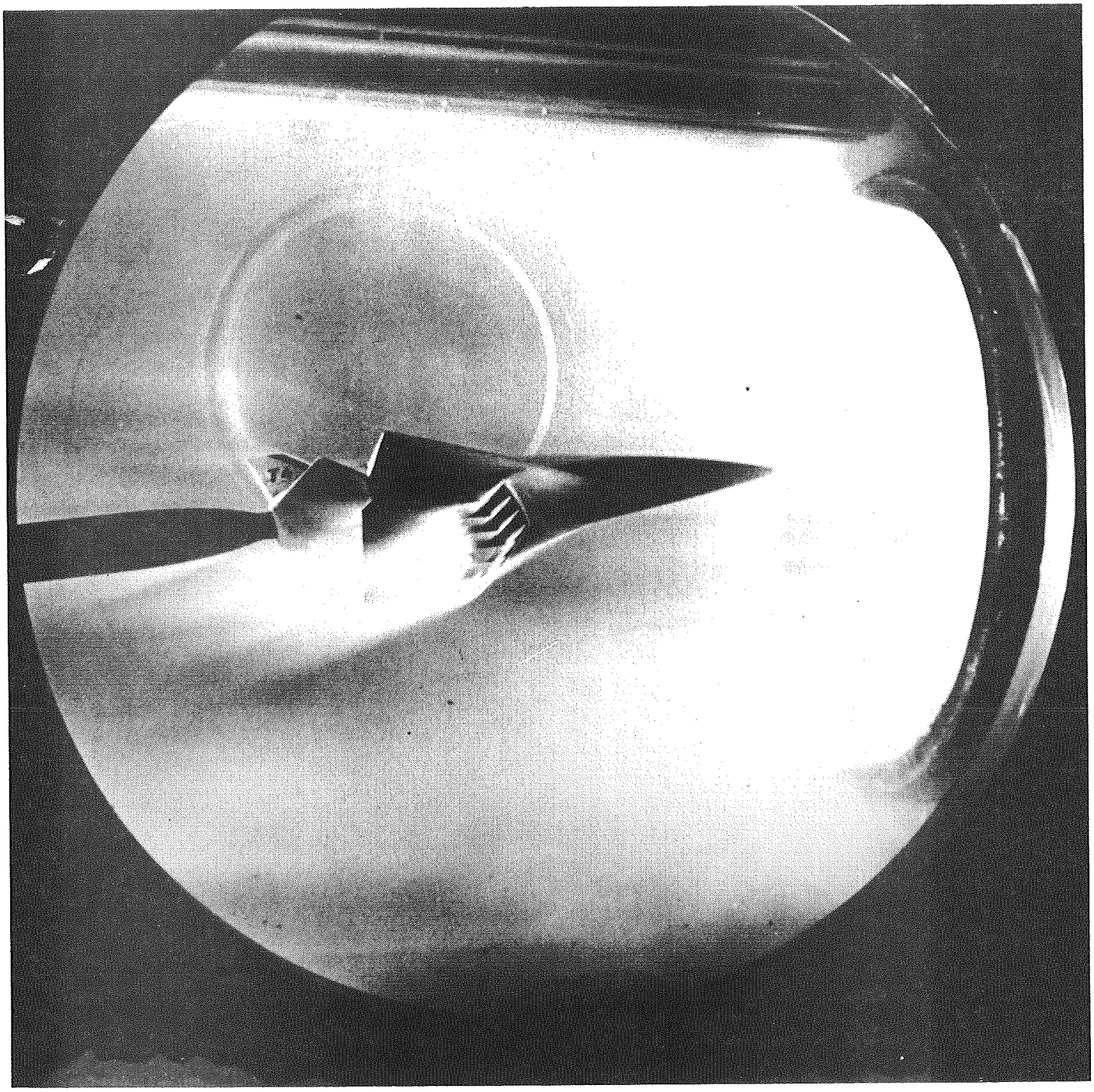


Rockwell International
Rocketdyne Division

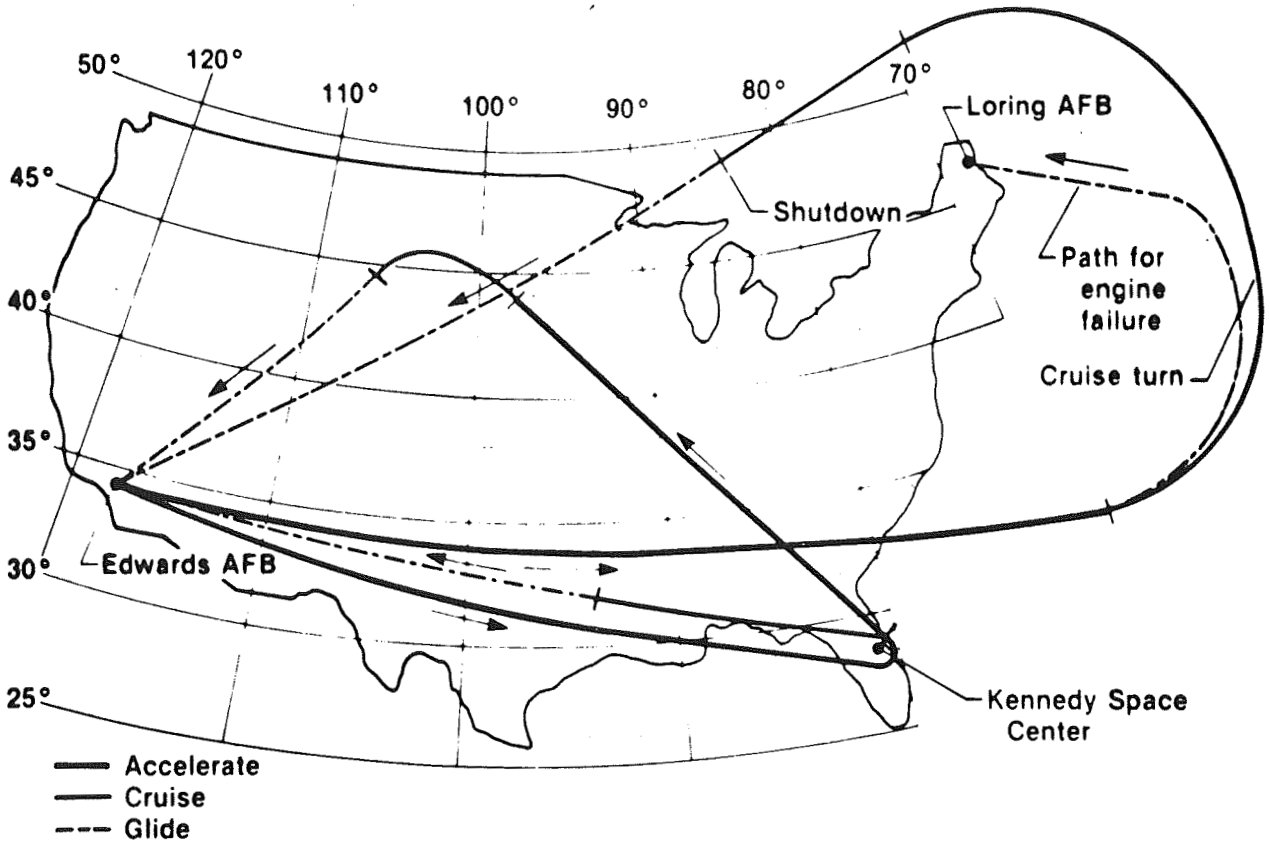
UNCLASSIFIED

NASP
ROCKETDYNE

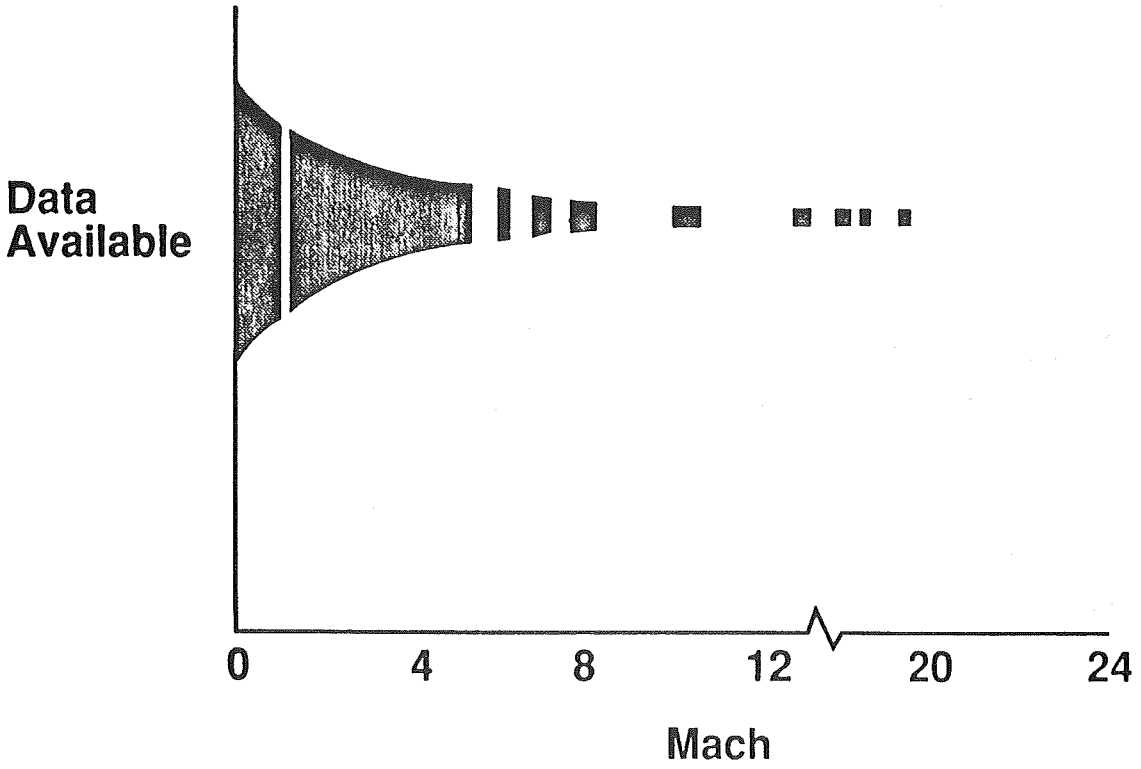




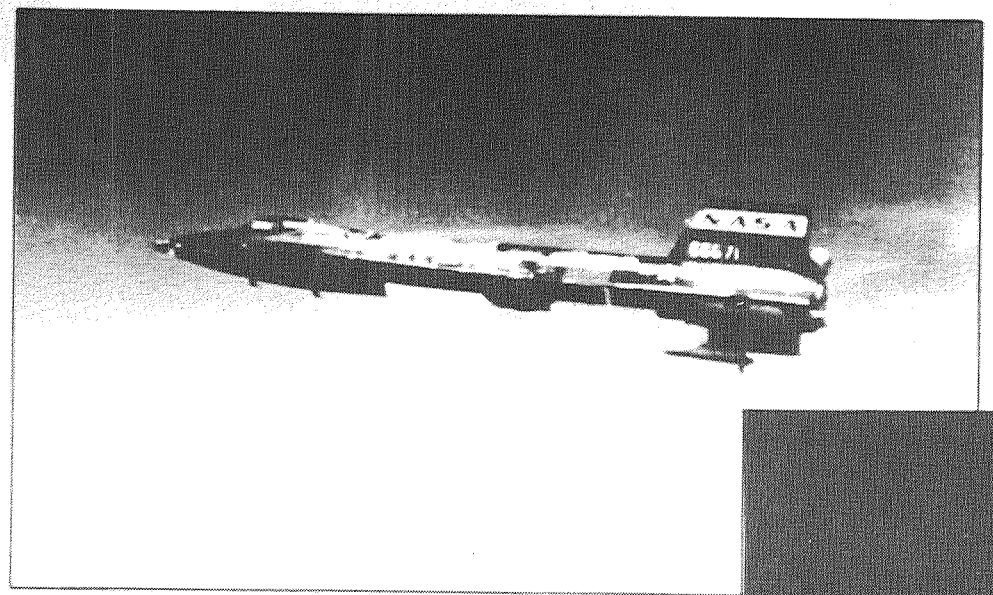
Ground Track for Envelope Expansion (U)



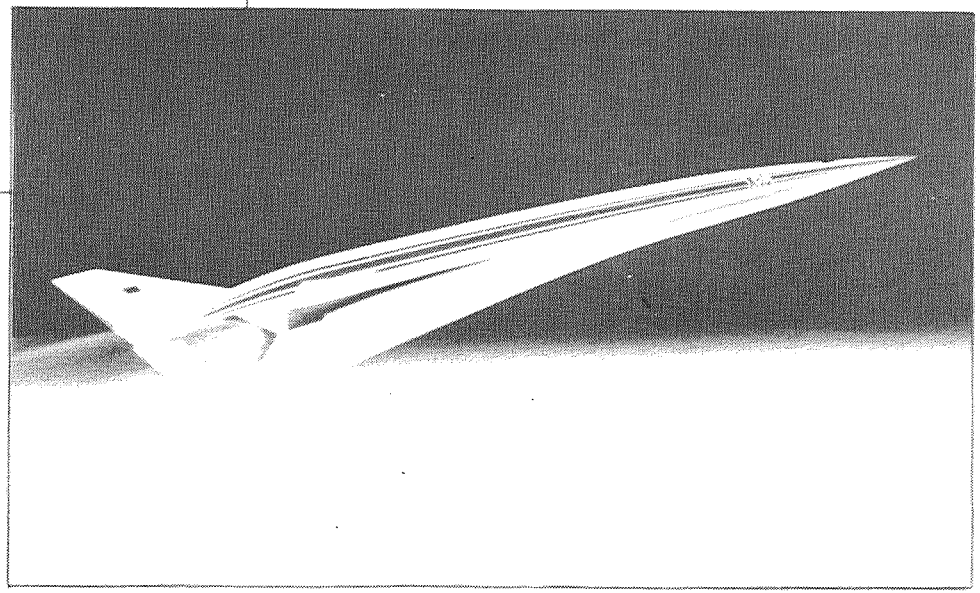
WIND TUNNEL DATA AVAILABLE FOR NASP (U)



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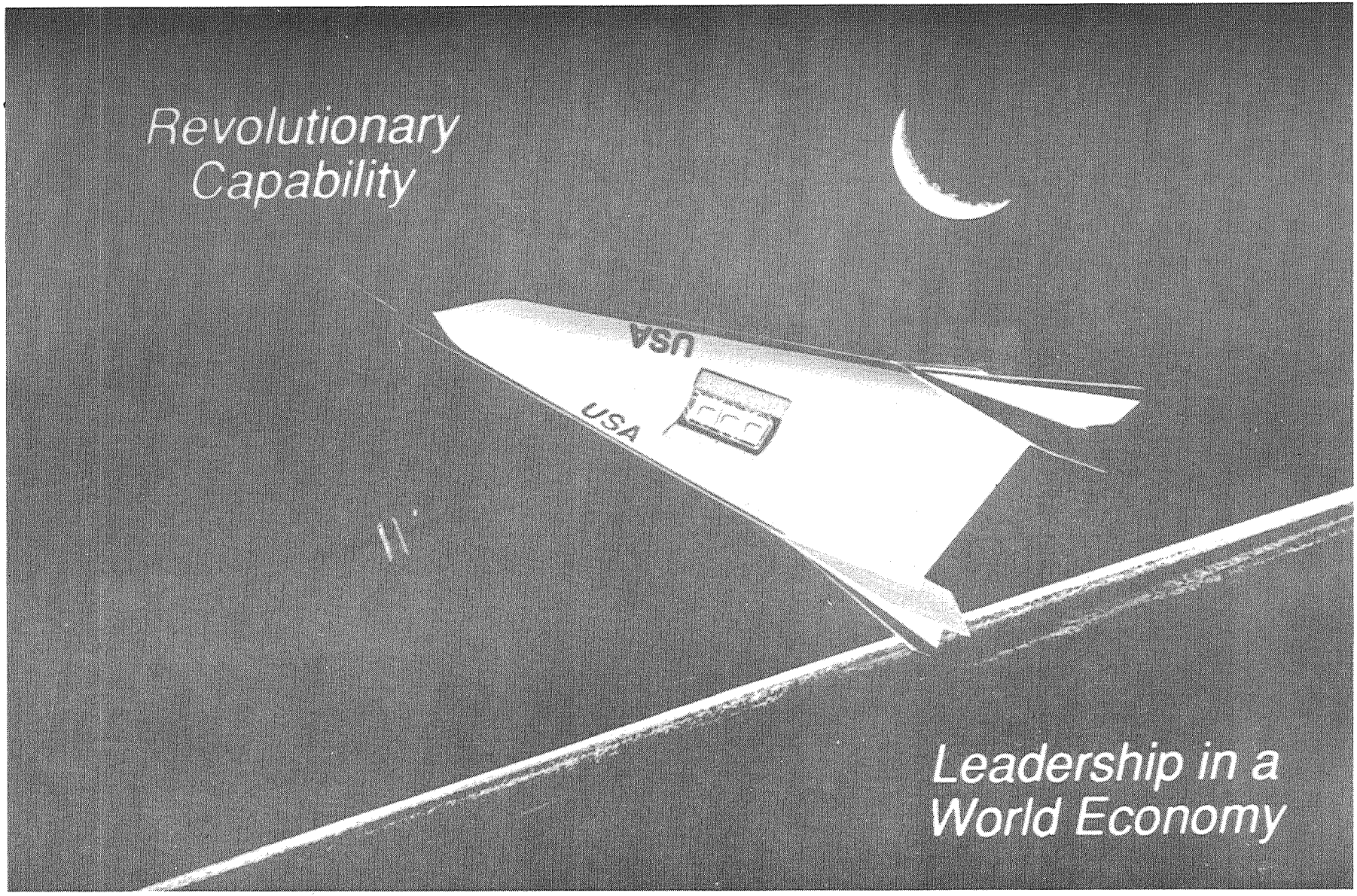


X-30

NASA

OAST

RN87-493(3)


Revolutionary
Capability
USA
USA
Leadership in a
World Economy

PROPULSION SYSTEMS OPTIONS-

FUTURISTIC SYSTEMS

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PRESENTATION 1.4.1

NUCLEAR AND SOLAR ELECTRIC PROPULSION

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SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM

FUTURISTIC SYSTEMS

SOLAR & NUCLEAR ELECTRIC PROPULSION

DAVE BYERS

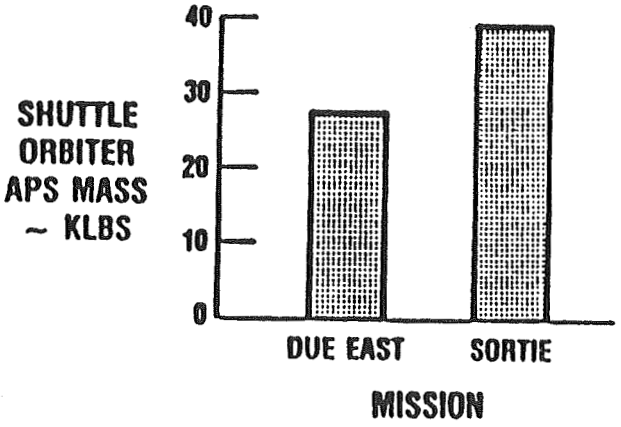
NASA LeRC

JUNE 27, 1990

AGENDA

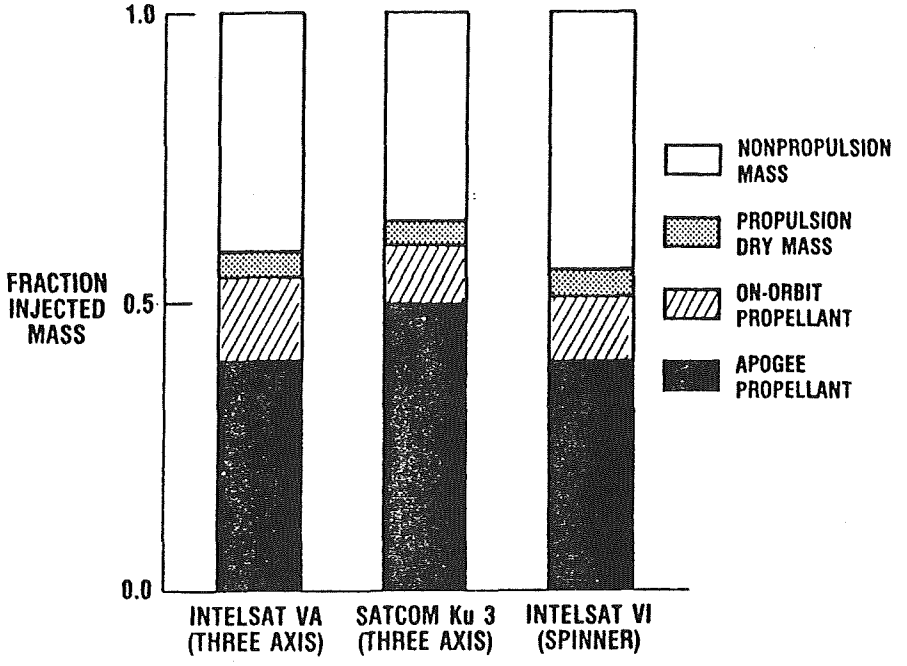
- IN-SPACE PROPULSION IMPACTS
- ELECTRIC PROPULSION
 - CHARACTERISTICS
 - CONSTRAINTS
 - CONCEPTS
 - STATUS
- MISSION IMPACTS OF ELECTRIC PROPULSION
- SUMMARY

APS OFFERS MAJOR LEVERAGE

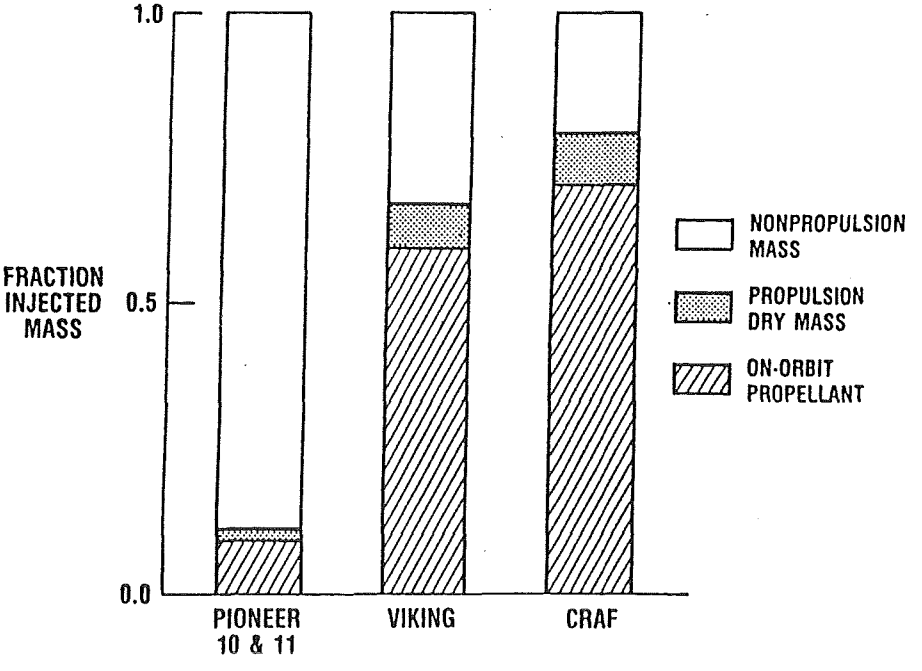


- APS MASS IS 11.4% TO 18.6% OF ORBITER

GEOSYNCHRONOUS TRANSFER ORBIT MASS FRACTIONS FOR RECENT COMMUNICATIONS SATELLITES



PLANETARY SPACECRAFT INJECTED MASS FRACTIONS



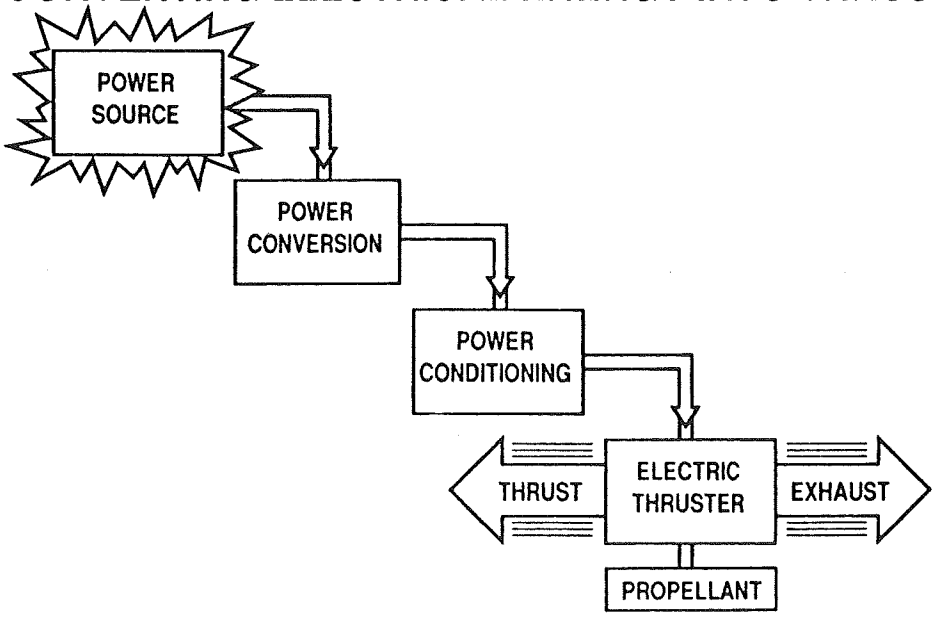
IN-SPACE PROPULSION IMPACTS

- PREDOMINANT MASS OF PAYLOADS NOW DELIVERED BY ETO AND ST VEHICLES
 - 12-19% OF ORBITER
 - 55-65% OF GTO
 - 70-80% OF PLANETARY
- IN-SPACE PROPULSION FRACTIONAL IMPACTS WILL INCREASE WITH INCREASED MISSION OBJECTIVES:
 - "DELTA V"
 - DURATIONS
 - PAYLOADS

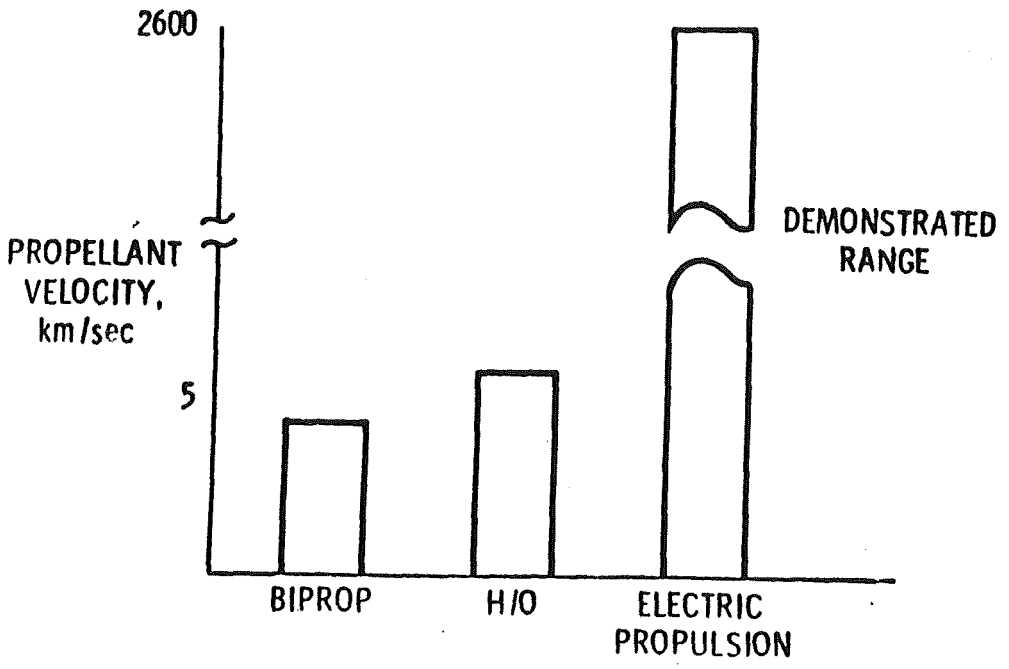
IN-SPACE PROPULSION IMPACTS

NON-INCREMENTAL IMPROVEMENTS IN FRACTIONAL IMPACT OF IN-SPACE PROPULSION WILL REQUIRE MAJOR TECHNOLOGY DELTA'S

CONVERTING ELECTRICAL ENERGY INTO THRUST



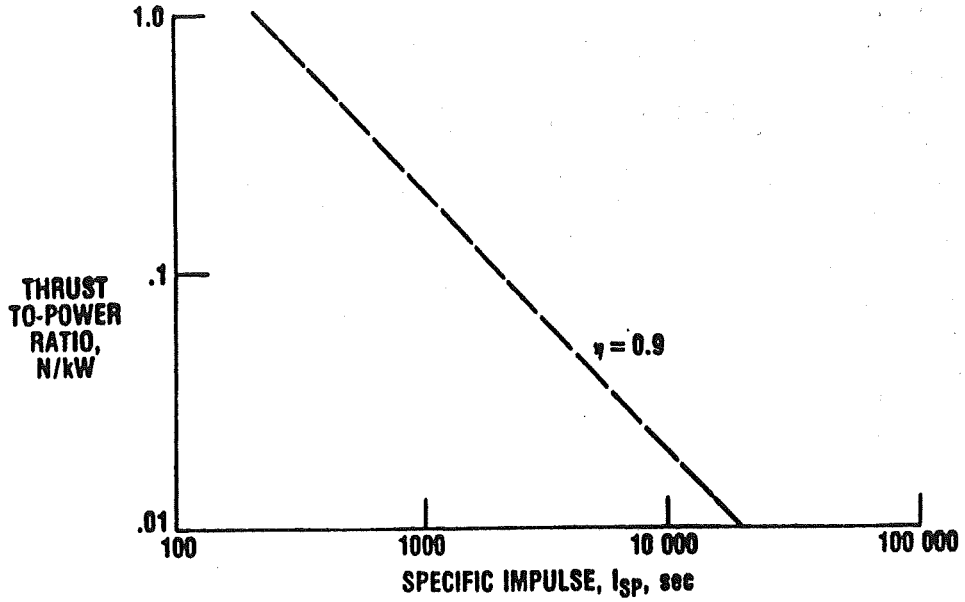
ELECTRIC PROPULSION ENABLES PROPELLANT VELOCITIES BEYOND FUNDAMENTAL LIMITS OF CHEMICAL SYSTEMS



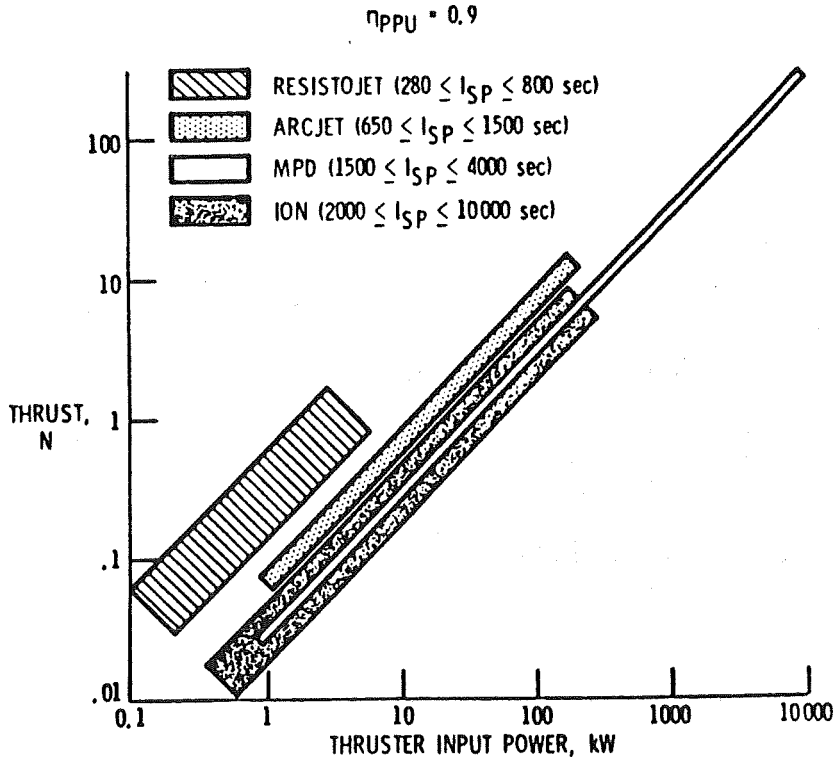
Electric Propulsion

The limits of achievable propellant velocity of chemical propulsion systems are shown along with the range of propellent velocities demonstrated with electric propulsion. With storable and H/O propellant the achievable upper values of propellant velocities are about 3750 and 5000 m/sec, respectively. The upper limit electric propulsion reflects tests in 1970 of a hydrogen ion thruster operated at about 70 KV. There is no fundamental subrelativistic propellant velocity with electric propulsion. The ability to achieve high propellant velocities is the fundamental charateristic of electric propulsion which has led to extensive R&T programs in a number of countries. Other key characteristics, which can potentially be of extreme benefit, are low thrust, precise impulse bit control (a quality exploited on the Navy NOVA satellites for precise ephemeris control (orbit constant to 0.01 seconds) and many propellant options including earth storables and inert gases.

THRUST-TO-POWER RATIO VS SPECIFIC IMPULSE



THRUST VS POWER FOR ELECTRIC PROPULSION



ELECTRIC PROPULSION

AFTER EQUATIONS ARE MASSAGED

$$\frac{\text{THRUST}}{\text{POWER}} = \frac{2\eta}{g I_{sp}}$$

THEREFORE!

- INCREASED FUEL EFFICIENCY (Isp) GAINED AT EXPENSE OF REDUCED THRUST FOR A GIVEN POWER
- ELECTRIC PROPULSION INHERENTLY RESULTS IN LOW ACCELERATIONS AND IS USEFUL ONLY FOR IN-SPACE APPLICATIONS

ELECTRIC PROPULSION

POWER OPTIONS

SEPS

- PHOTOVOLTAIC
 - SPACE STATION FREEDOM
 = 40W/KG (@ 1AV)
 - APSA
 = 130w/kg (@ 1AV)

NEPS

- SP-100 REACTOR + CONVERSION
 - STATIC (TO 100 KW)
 = 30W/KG
 - DYNAMIC (GROWTH)
 = ⩾ 100 W/KG

- SPACE STATION FREEDOM PROGRAM PROVIDING LARGE ARRAY EXPERIENCE AND INFRASTRUCTURES
- APSA PROJECT DEVELOPING LIGHTWEIGHT ARRAYS
- SP-100 REACTOR ONLY ACTIVE POWER REACTOR PROGRAM
 - STATIC & DYNAMIC CONVERSION TECHNOLOGIES UNDER DEVELOPMENT

ELECTRIC PROPULSION

CONCEPT SUMMARY

ELECTRIC PROPULSION

THREE CLASSES OF CONCEPTS

ELECTROTHERMAL

- GAS-HEATED BY RESISTORS AND/OR ARCS AND EXPANDED THROUGH A NOZZLE
 - RESISTOJETS
 - ARCJETS
 - PULSED

ELECTROSTATIC

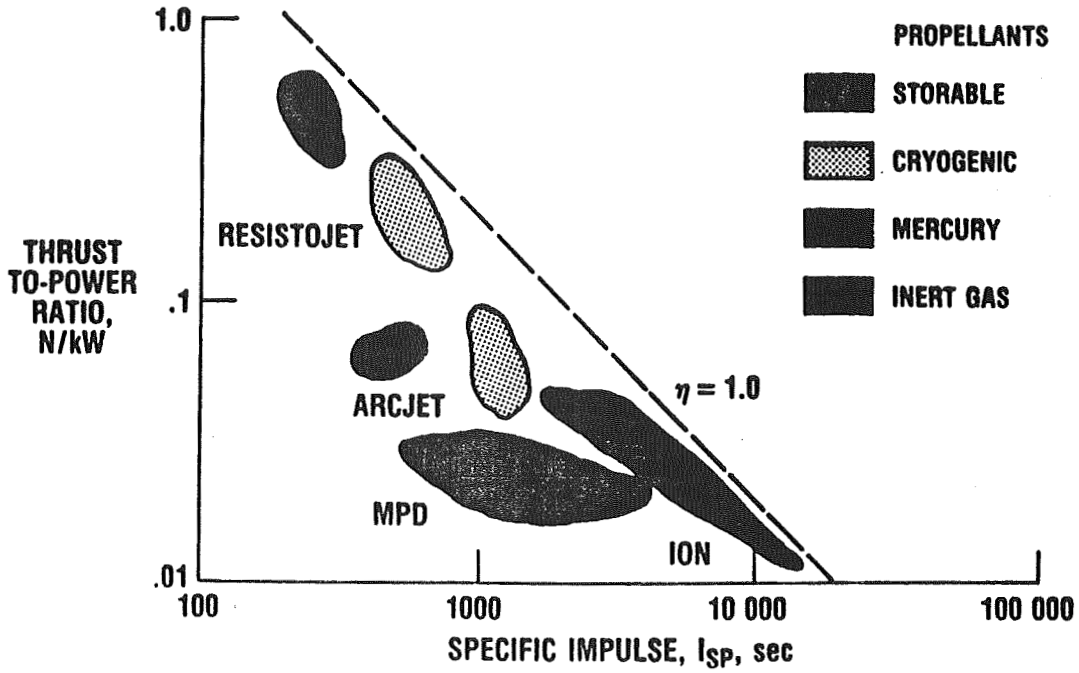
- IONS ELECTROSTATICALLY ACCELERATED
 - ION

ELECTROMAGNETIC

- PLASMAS ACCELERATED BY ELECTRIC AND MAGNETIC FIELDS
 - MPD
 - PULSED PLASMA

THRUST-TO-POWER RATIO VS SPECIFIC IMPULSE

$\eta_{PPU} = 0.9$



ELECTRIC PROPULSION

STATUS

ELECTRIC PROPULSION

STATUS

77 SPACE TESTS CONDUCTED

TYPE		ORIGIN	
ELECTROTHERMAL	33	CHINA	1
ELECTROSTATIC	16	JAPAN	7
ELECTROMAGNETIC	28	USSR	21
		USA	48
	77		77

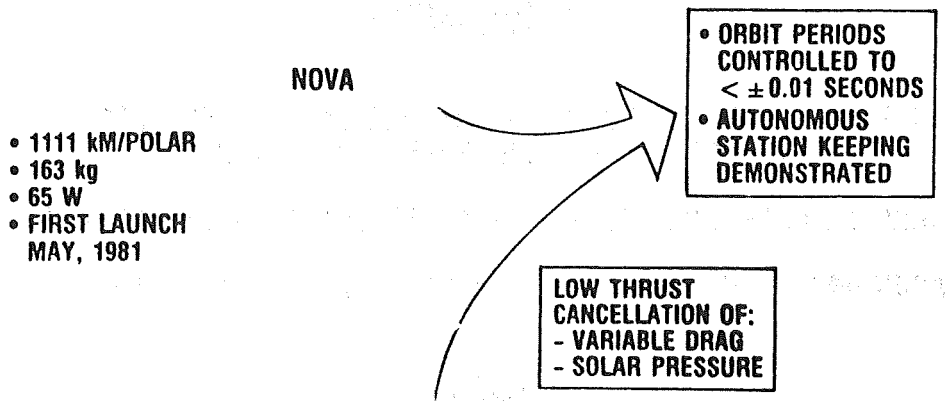
(1) SCHREIB, R., AIAA PAPER NO. 88-0777, MARCH 1988

STATUS

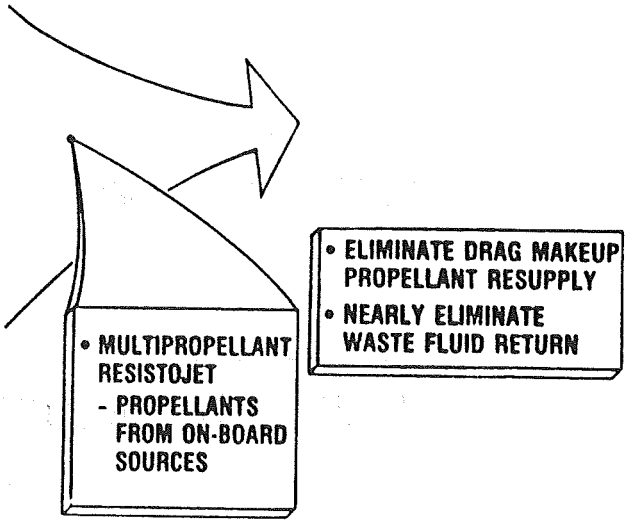
LOW POWER (ORBIT ADJUST) SYSTEM OPERATIONAL/BASELINED

- NOVA
- US COMMUNICATION SATELLITES
- SPACE STATION

ELECTRIC PROPULSION PROVIDES PRECISION ORBITS FOR NOVA SATELLITES



ELECTRIC PROPULSION SELECTED FOR IOC SPACE STATION DRAG MAKEUP PROPULSION



ELECTRIC PROPULSION STATUS SUMMARY

PROPULSION

- MANY CONCEPTS EVALUATED AND FLIGHT TESTED
 - ONLY TWO SPACE TESTS OVER 1KW
- LOW POWER SYSTEMS OPERATIONAL AND BASELINED
- PRIMARY PROPULSION CONCEPTS IN R&T PHASES

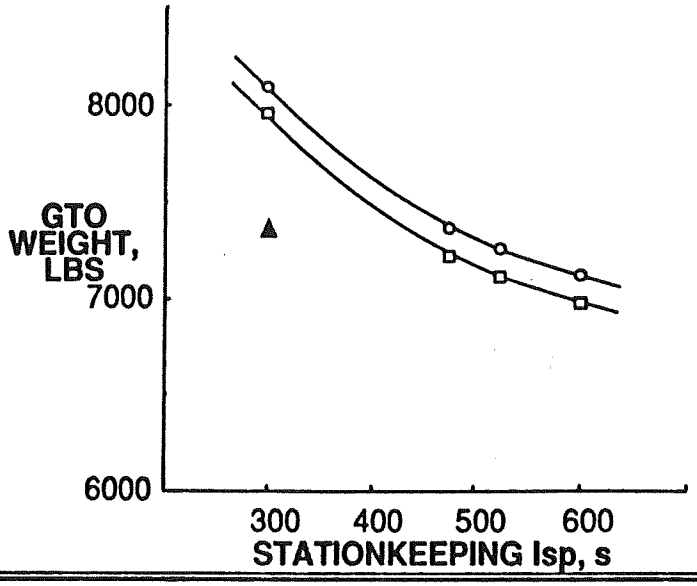
POWER

- SAFE ARRAY STRUCTURE DEMONSTRATED IN SPACE
- LIGHTWEIGHT ARRAYS UNDER DEVELOPMENT
- SP-100 REACTOR PROGRAM ON-GOING
- MULTI-MEGAWATT REACTORS UNDER STUDY

ELECTRIC PROPULSION

MISSION IMPACTS

ON-BOARD PROPULSION IMPACTS(1)

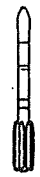


ADVANCED STATIONKEEPING AND APOGEE PROPULSION

- **REDUCED GTO REQUIREMENTS**
- **MITIGATED LAUNCH SITE IMPACTS**
- **INCREASED LEVERAGE FOR LONG LIFE SATELLITES**

(1) 15 YEAR GEO LIFE, 3500 LBS EOL WEIGHT

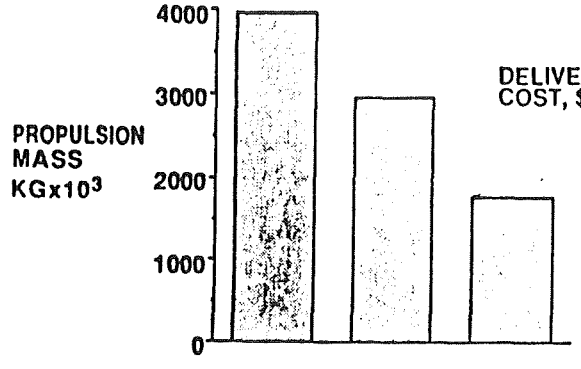
ADVANCED ORBIT TRANSFER PROPULSION IMPACTS(1)

	ELECTRIC	CHEMICAL
		
MLEO, Lbs	10307	37782
TRIP TIME, DAYS	180	1
LAUNCHER	DELTA II	TITAN IV
OTV	SEPS	IUS

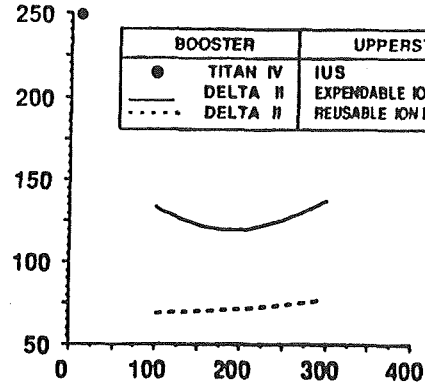
ELECTRIC PROPULSION OFFERS 3X MLEO REDUCTION

(1) AIAA 89-2496 "Electric Orbit Transfer Vehicle - A Military Perspective", S. Rosen and J. Sloan /AFSD. 5250 Lbs to GEO

LEO→MEO→LEO (1)



DELIVERY COST, $M



LEO TO GEO (2)

BOOSTER	UPPERSTAGE
TITAN IV	IUS
DELTA II	EXPENDABLE ION EOTV
DELTA II	REUSABLE ION EOTV

PROPULSION	N_2H_4	NTO/N_2H_4	ARCJET
ISP,S	220	310	500
TRANSFER TIME,DAYS	0.2	0.2	40
POWER,KW	--	--	5

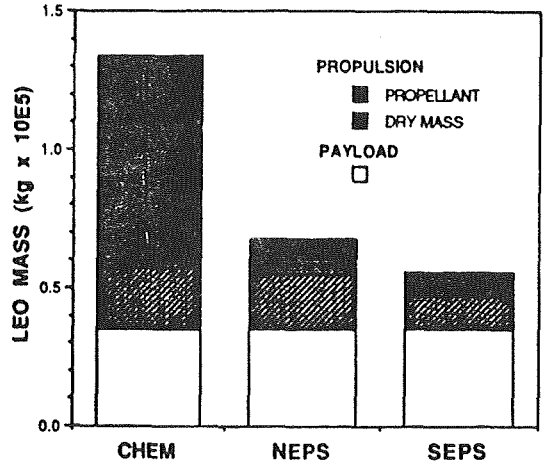
ADVANCED PROPULSION OFFERS GREAT BENEFITS FOR E-O FREE FLYERS

(1) EARTH OBSERVATION SYSTEM
(DRY MASS, $9x10^3$ KG)
(POWER, 8 kW)

(2) DATA FROM AEROSPACE CORP.
(PAYLOAD, 2380 KG)

CARGO VEHICLE PROPULSION

LUNAR MISSION (1)
(LEO → LMO → LEO)

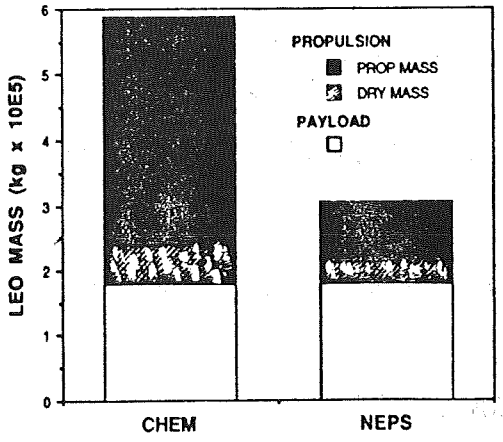


PROPULSION	CHEM	NEPS	SEPS
o Power, MW	--	1	0.3
o Trip Time, Days	10	257	769
o Aerobraking	Yes	No	No

ELECTRIC PROPULSION PROMISES SIGNIFICANT LEO MASS REDUCTIONS

CARGO VEHICLE PROPULSION

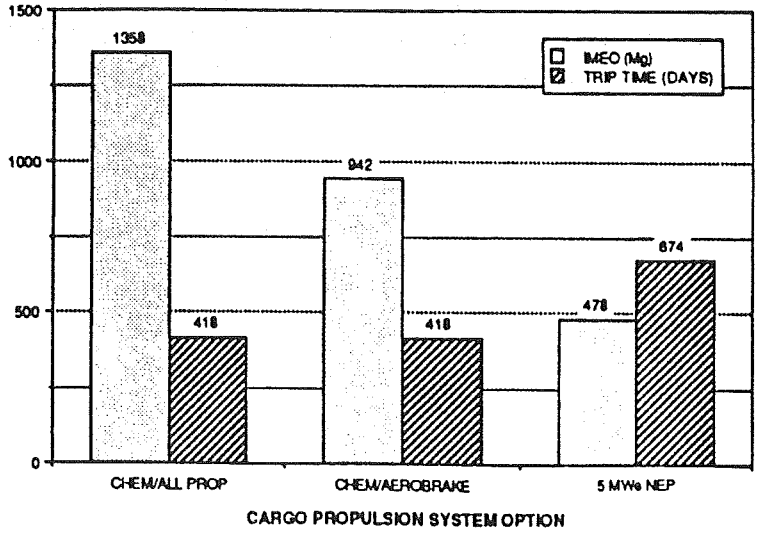
MARS MISSION
(LEO → LMO)



PROPULSION	CHEM	NEPS
o Power, MW	--	4
o Trip Time, Days	210	700
o Aerobraking	Yes	No

ELECTRIC PROPULSION PROMISES SIGNIFICANT LEO MASS REDUCTIONS

NASA LEWIS RESEARCH CENTER



IMEO and One Way Trip Time Comparison of Potential Transportation Systems for Mars Cargo Delivery. Vehicles Deliver 222 Mg from LEO to Phobos.

HUMAN EXPLORATION INITIATIVE OFFICE

SOLAR & NUCLEAR PROPULSION

SUMMARY

ETO & ST VEHICLE PAYLOADS OFTEN PREDOMINATELY IN-SPACE PROPULSION

- **MITIGATED SIGNIFICANTLY ONLY BY NEW IN-SPACE PROPULSION**

ELECTRIC PROPULSION STATUS

- **LOW POWER APPLICATIONS IN PLACE AND GROWING**
- **HIGH POWER APPLICATIONS REQUIRE PROPULSION AND POWER DEVELOPMENTS**

ELECTRIC PROPULSION IMPACTS:

- **1000 LBS GTO REDUCTIONS**
- **2 TO 3X REDUCTIONS IN MLEO FOR MAJOR MISSIONS**
- **TRIP TIME PENALTY/BENEFITS VERY MISSION SPECIFIC**
- **GREAT EXPANSIONS OF LAUNCH WINDOWS**

PRESENTATION 1.4.2

NUCLEAR THERMAL PROPULSION

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NASA

Nuclear Thermal Propulsion

Space Transportation Propulsion Technology Symposium

Gary L. Bennett

Program Manager
Propulsion, Power and Energy Division
Office of Aeronautics, Exploration and Technology
27 June 1990

DIRECT FISSION-THERMAL PROPULSION PROCESS

NUCLEAR ENERGY

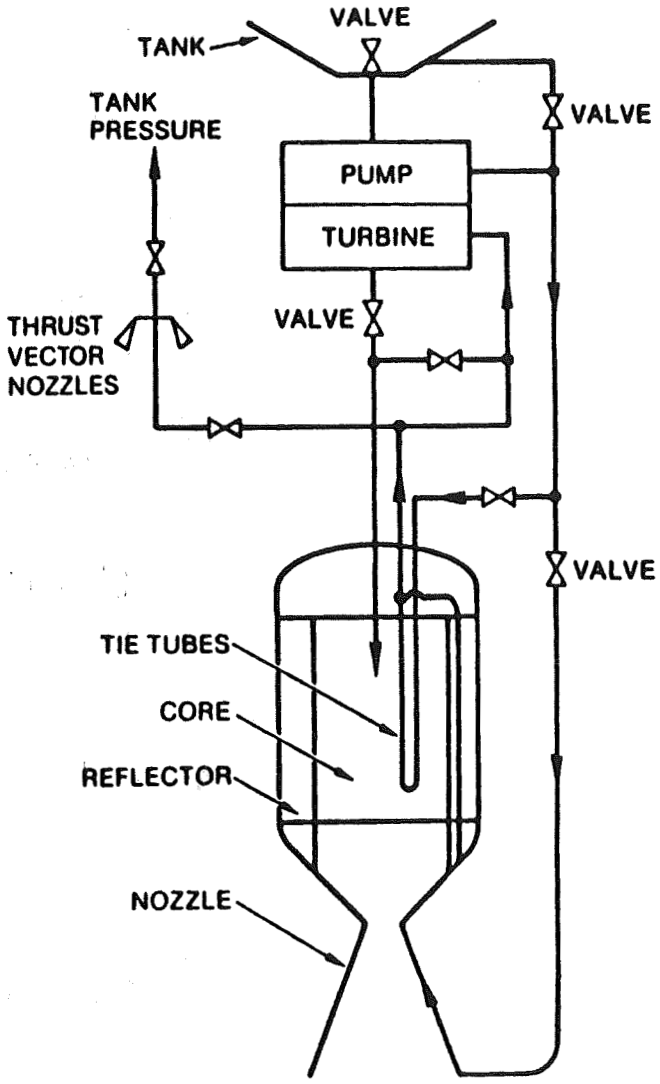
FRAGMENT KINETIC ENERGY

THERMAL ENERGY

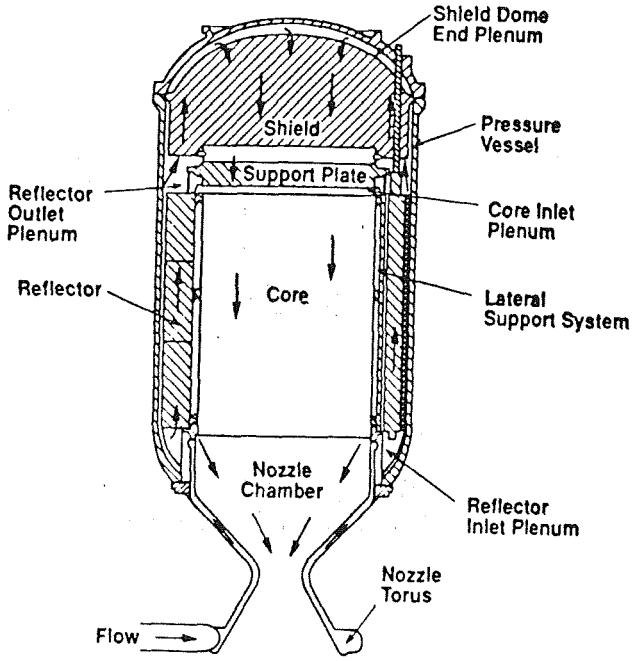
THERMODYNAMIC EXPANSION

DIRECTED THRUST

NUCLEAR ENGINE SCHEMATIC



Typical Rocket Propulsion Reactor

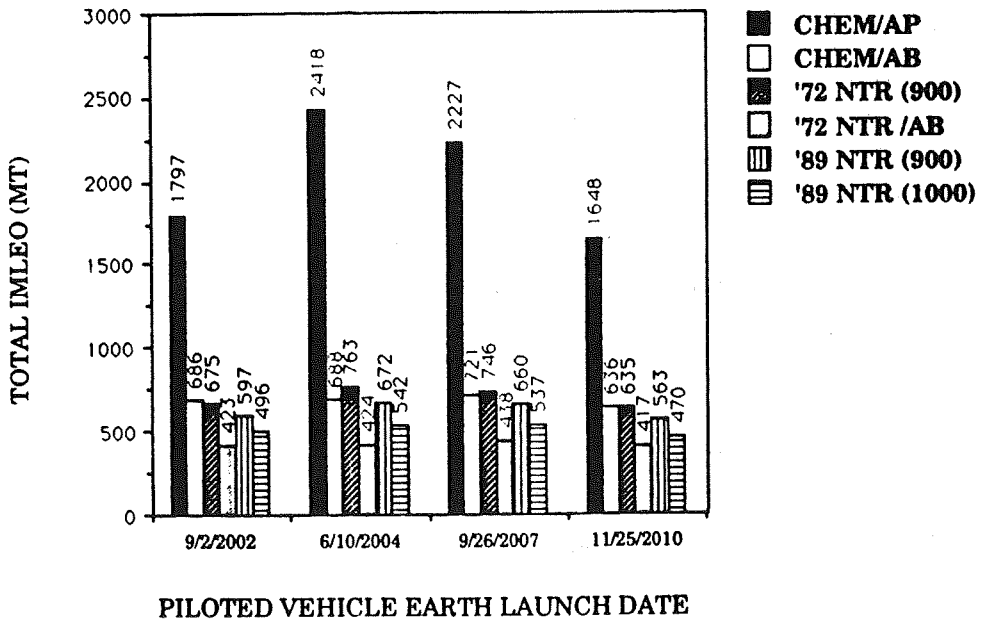


DIRECT FISSION-THERMAL PROPULSION -- ADVANTAGES

- HIGH SPECIFIC IMPULSE (860s - 1000s)
- HIGH THRUST-TO-WEIGHT
- NOT ENERGY LIMITED (AUX. POWER OPTIONS)
- REUSABLE
- THROTTLEABLE (25% - 100%)
- MONO-PROPELLANT
- MULTIPLE PROPELLANT CHOICES (H_2, NH_3, . . .)
- NEAR-TERM TECHNOLOGY (IT WORKS!)

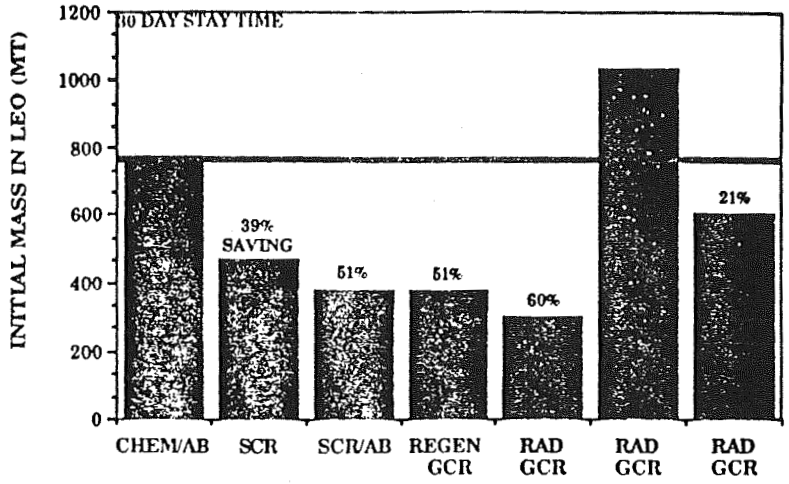
MISSION APPLICATIONS OF DIRECT FISSION-THERMAL PROPULSION

- ORBIT TRANSFER VEHICLE
- LUNAR TUG
- PILOTED MARS MISSION
- EXTRA-TERRESTRIAL RESOURCE UTILIZATION



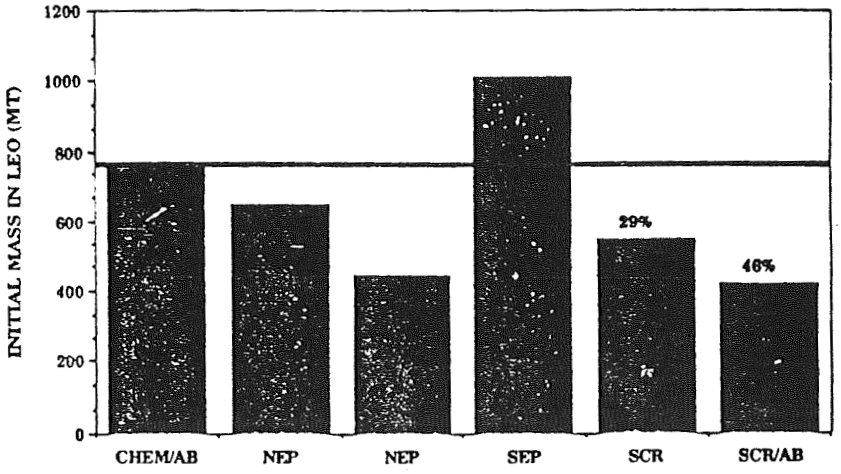
MARS EXPEDITION CASE - IMLEO SENSITIVITY TO LAUNCH OPPORTUNITY

PROPULSION PERFORMANCE COMPARISON
SCR AND GCR PILOTED MARS MISSIONS, QUICK TRIPS



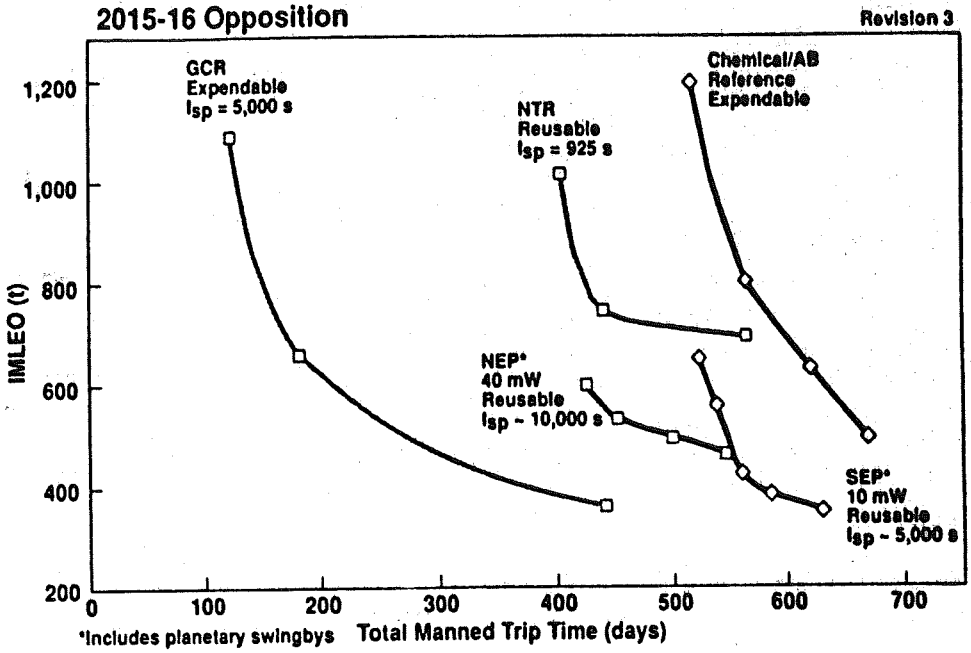
Isp (s)	475	1000	1000	2500	5000	5000	5000
Trip Time (d)	434	434	434	434	434	120	180

PROPULSION PERFORMANCE COMPARISON
NEP, SEP, AND SCR PILOTED MARS MISSION



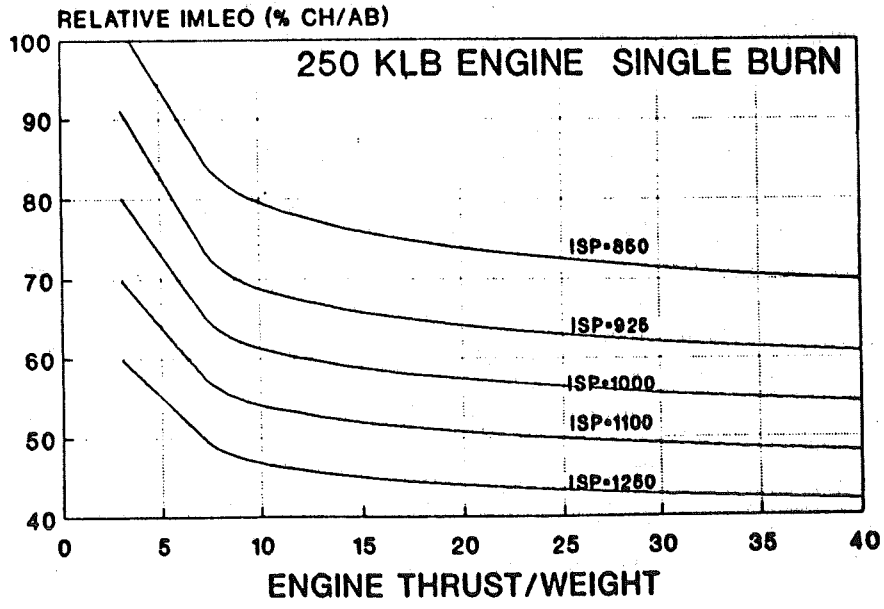
Alpha (kg/kWe)	N/A	15	10	10	N/A	N/A
Isp (s)	475	5000	5000	5000	900	900
Trip Time (d)	434	650	650	650	434	434

Various Opportunities For Given MTV Propulsion Options

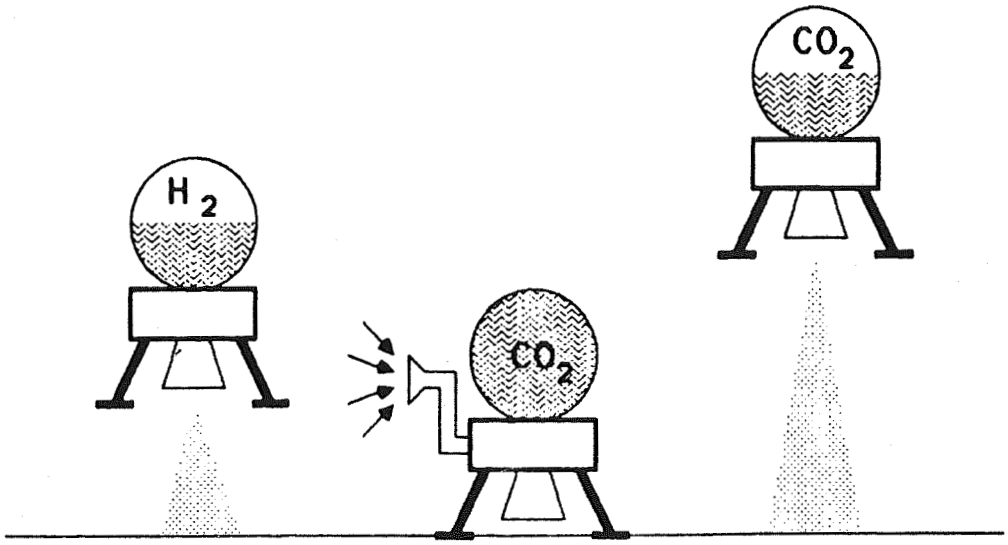


Ref: Boeing - Advanced Civil Space Systems

NTR MARS PERFORMANCE THRUST/WEIGHT AND ISP VARIATIONS



EXTRA-TERRESTRIAL PROPELLANT LANDER/HOPPER/ASCENT VEHICLE (DIRECT FISSION-THERMAL PROPULSION)



Land on Mars with Hydrogen

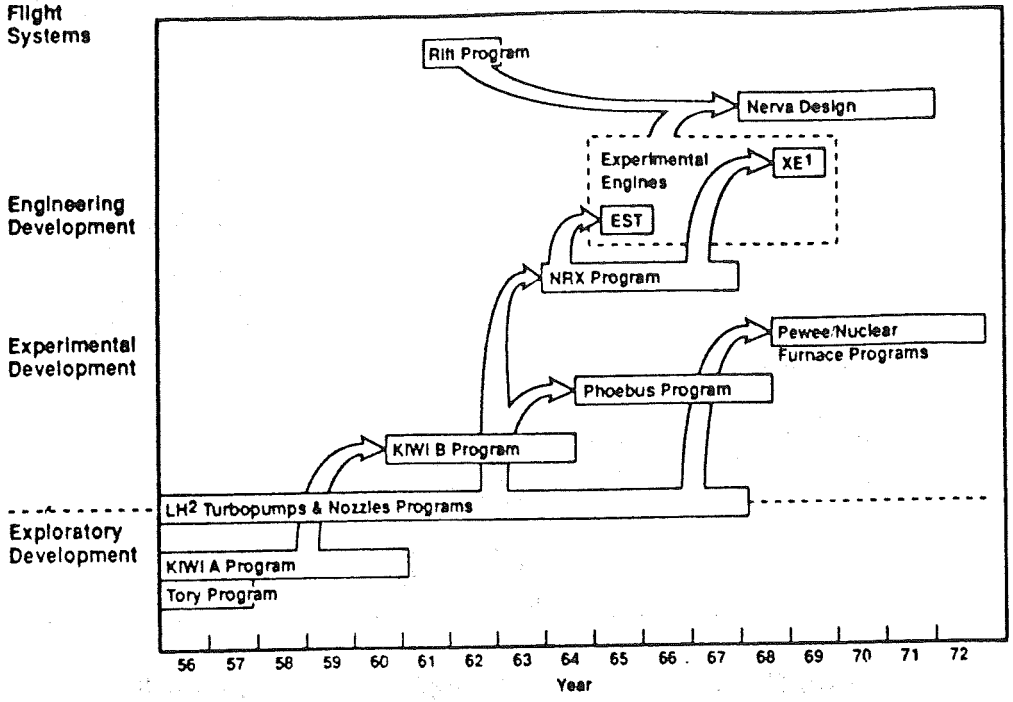
Refuel with CO_2

Launch with CO_2

DIRECT FISSION-THERMAL PROPULSION SYSTEMS

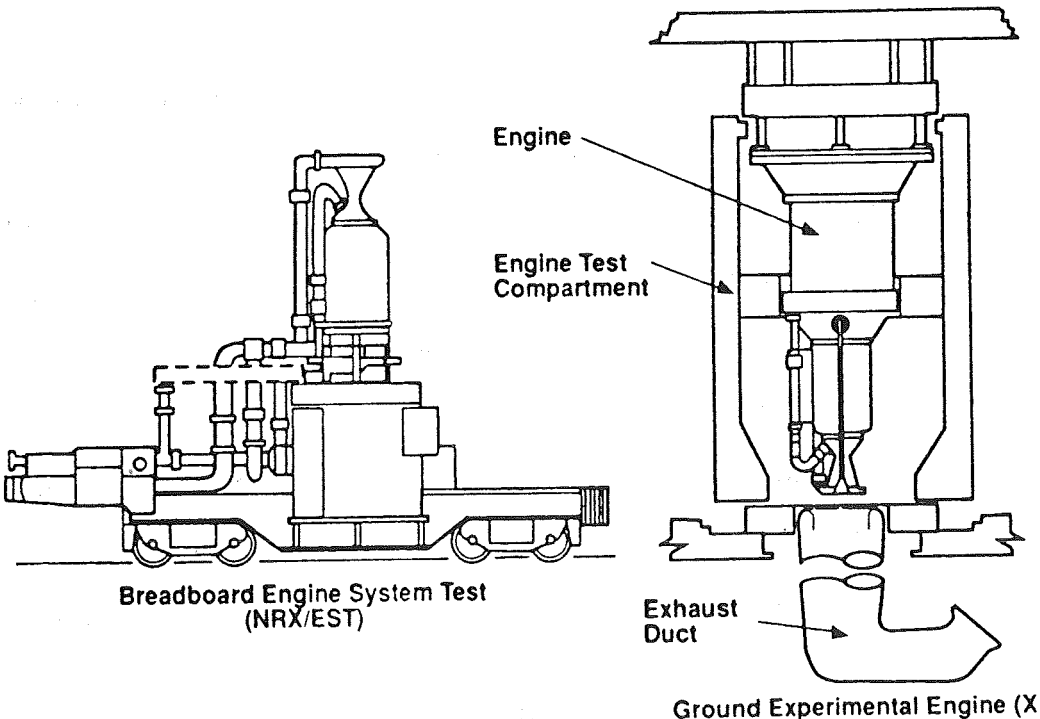
	NERVA	ANRE (INEL)	PBR (BNL)
THRUST [kN]	333	65	44
I_{sp}[s]	825	900	900-1000
MASS FLOW [Kg/s]	41.2	7.4	4.5
POWER [MW_T]	1500	370	200
WEIGHT [10^3Kg]	10.4	2.1	0.6
THRUST/WEIGHT	32	31	73
FUEL	UC	UC-ZrC-C	UC-ZrC
MODERATOR	C	ZrH	LIH, Be, ZrH

Nuclear Rocket Program



NERVA Engine Technology Testing

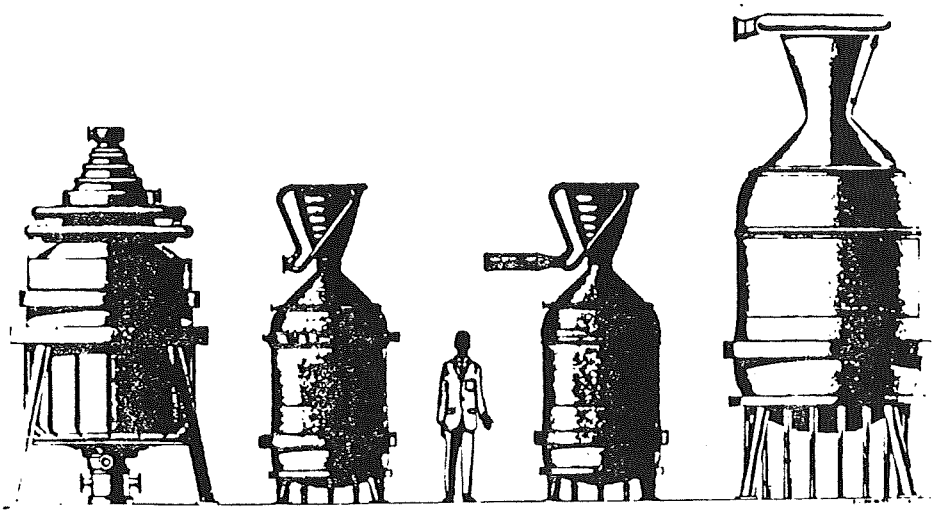
Test Stand Superstructure



What we are attempting to make is a flyable compact reactor, not much bigger than an office desk, that will produce the power of Hoover Dam from a cold start in a matter of minutes

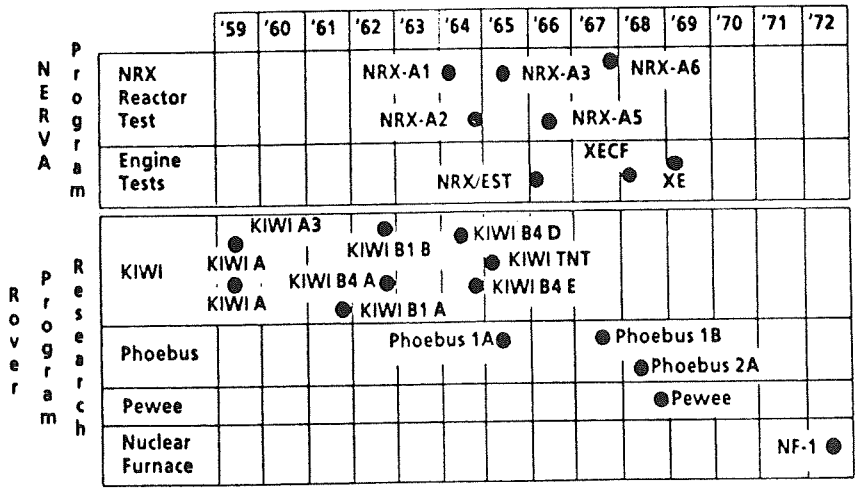
-- Dr. Glenn T. Seaborg
Chairman
Atomic Energy Commission

Evolution of Rover Reactors



KIWI A	KIWI B	Phoebus 1	Phoebus 2
1958-60	1961-64	1965-66	1967
100 Megawatts	1,000 Megawatts	1,000 & 1,500 Megawatts	5,000 Megawatts
0 lb Thrust	50,000 lb Thrust	50,000 lb Thrust	250,000 lb Thrust

NERVA/Rover Reactor System Test Sequence



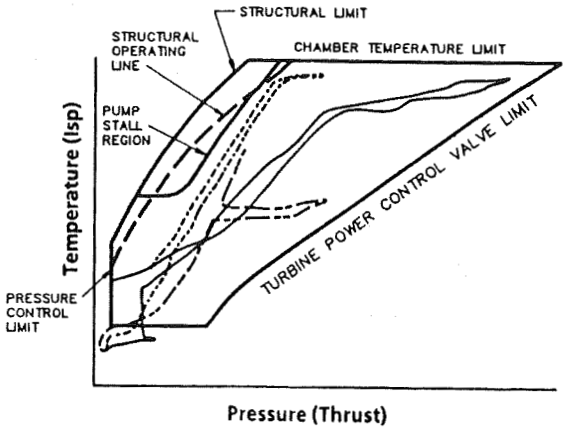
Flexibility Demonstrated in Ground Experimental Engine (XE) Test

TEST PERIOD, 3/20/69–8/28/69

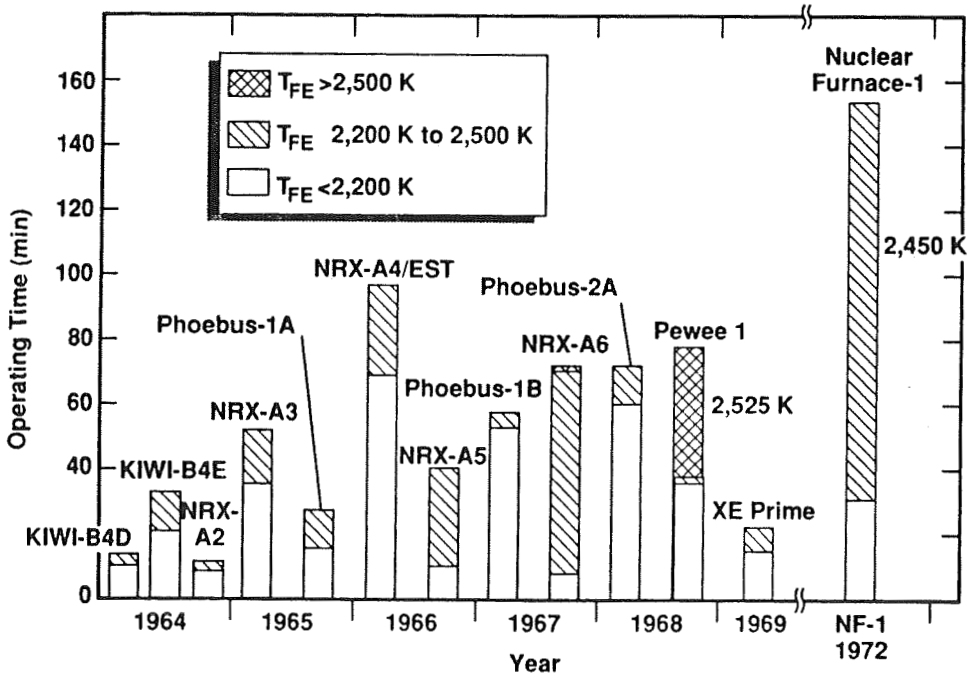
EXPERIMENTS CONDUCTED

STARTUP INVESTIGATIONS	15
PERFORMANCE CHARACTERISTICS AT HIGH POWER	6
ENGINE DYNAMIC PERFORMANCE	10
FACILITY EVALUATION	4

SUMMARY – 28 STARTUPS
3 HOURS 48 MINUTES OF OPERATION



Operating Time vs Temperature for Nuclear Rocket Program



Major System Test Results

- Demonstrated power capability
 - 1100 MWt in NRX (55,000 lbs thrust)
 - 4400 MWt in Phoebus (220,000 lbs thrust)
- Demonstrated power, temperature and flow stability
 - High specific impulse (800-900 seconds)
 - High thrust startup
- Demonstrated reactor/engine endurance – 60 minutes in NRX-A6
- Demonstrated reactor/engine maneuverability - 28 startup cycles in NRX-XE
- Demonstrated reactor fuel – 10 hours 40 minutes and 64 cycles

NERVA TECHNOLOGY HAS SYNERGISTIC APPLICATIONS

Steady-State Power

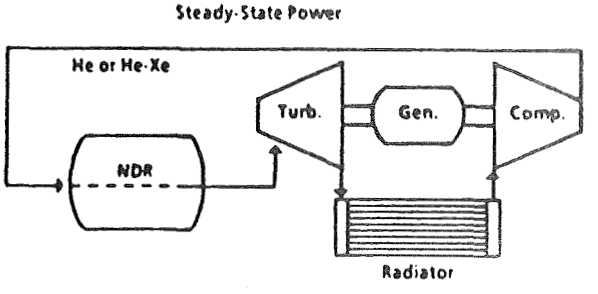
- 10's of MWe for electric propulsion

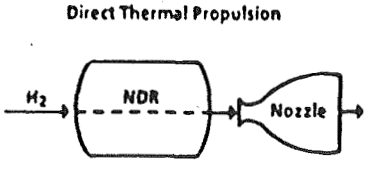
Direct thermal propulsion

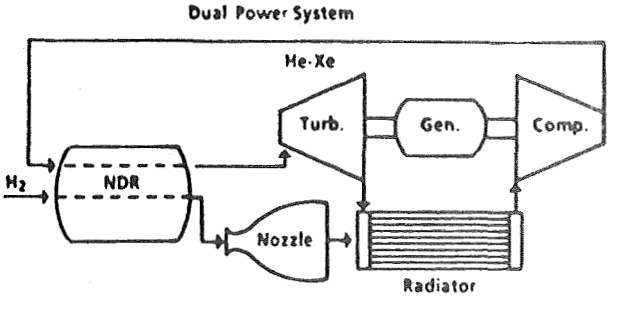
- 15,000 to 250,000 pounds of thrust

Dual Power Systems

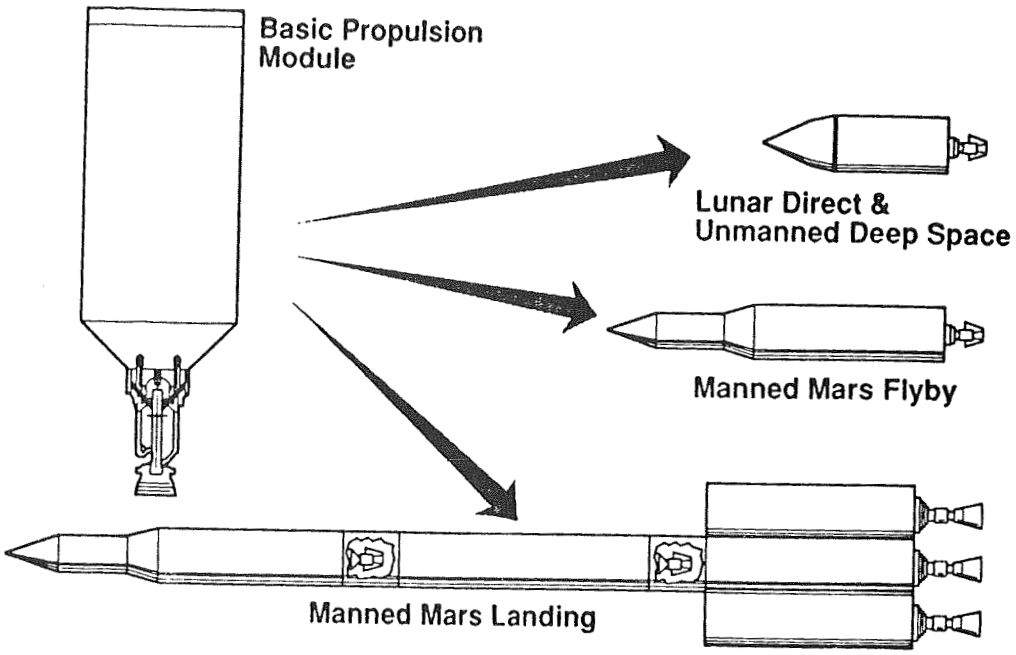
- High direct thrust (e.g., 75,000 pounds) plus low electric propulsion (e.g., 1MWe)







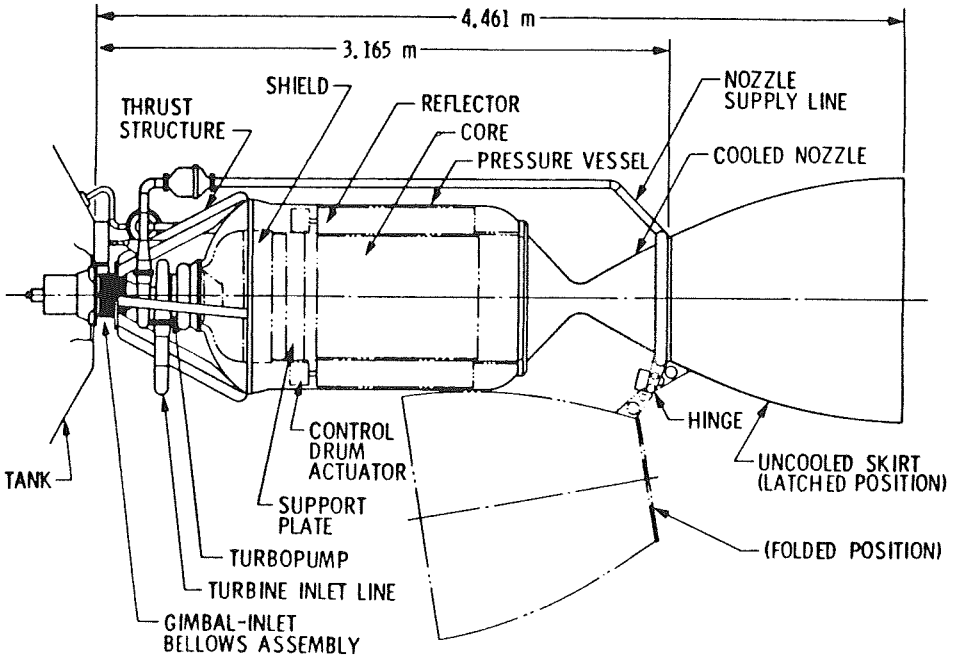
Versatility



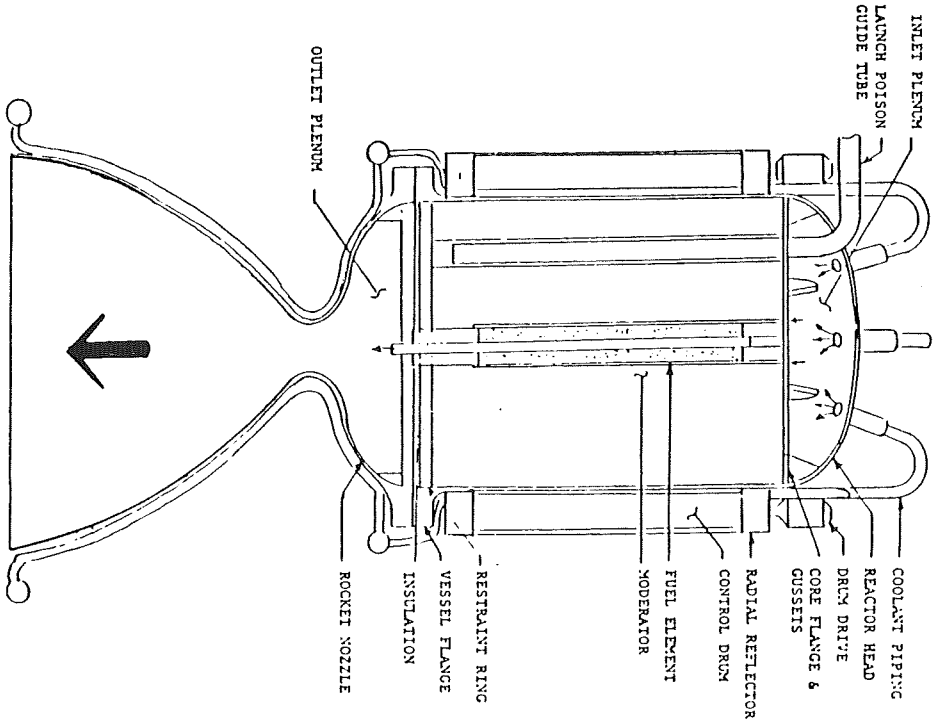
NERVA DESIGN - STATUS

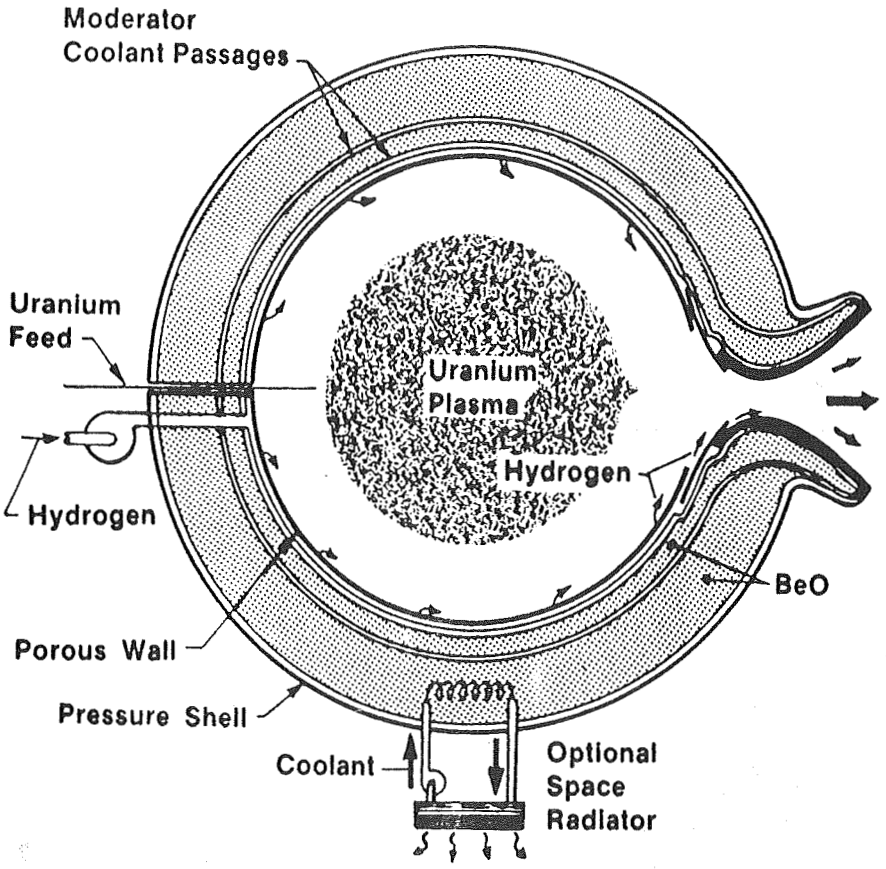
- DEVELOPED DURING PROJECT ROVER
- FULL POWER, FULL DURATION TESTED
- FLIGHT QUALIFIABLE DESIGN UNDER WAY AT CONCLUSION OF PROJECT ROVER
- EXPERTISE STILL AROUND (JUST BARELY)
- WESTINGHOUSE FULL-DESIGN BLUEPRINTS INTACT
- LANL CAN STILL EXTRUDE FUEL SEGMENTS
- FUEL SEGMENT TEST FACILITIES AVAILABLE
- FULL SCALE TEST FACILITIES UNAVAILABLE

SMALL/ADVANCED NUCLEAR ROCKET ENGINE
(SNRE/ANRE - LANL/INEL)



PARTICLE BED REACTOR DESIGN

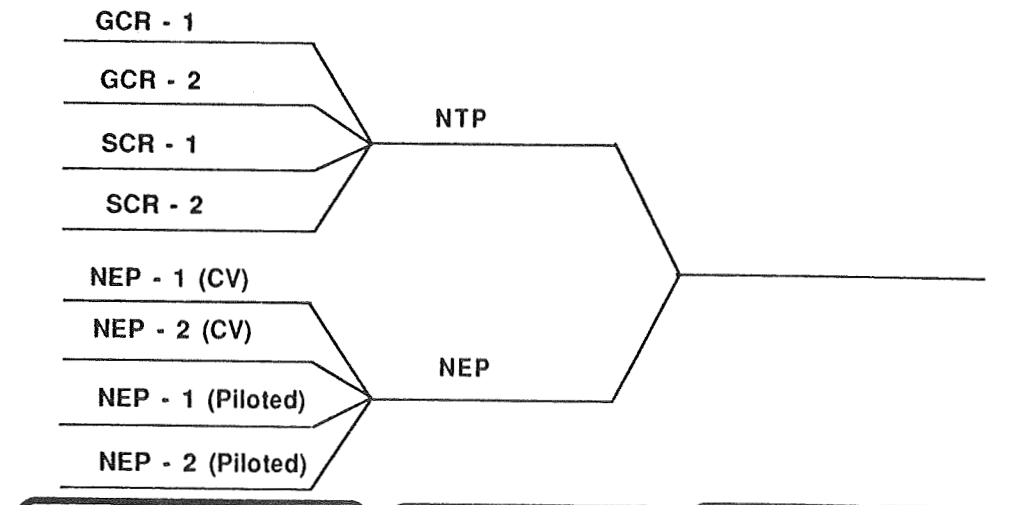




NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

5.3 TECHNOLOGY DEVELOPMENT STRATEGY



NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

EXECUTIVE SUMMARY

KEY TECHNICAL ISSUES

Safety/safeguards/QA
(during all program phases)
Qualification/acceptance test strat.
Reliability and fault tolerance
High Performance engines
(including reactors)
Reusability/restart capability
Reactor Fuel
Structural Aspects
Turbomachinery
Vessels/Nozzles
Pumps/Valves
Diagnostic Capability
Control Systems (neutronics/
I&C)
Power Processing Units
(NEP)
Thrusters (NEP)
Space operations
- radiation shielding
- design criteria for in-space
operation and
maintenance
Propellants/Prop. handling
Thermal hydraulics
Thermal Management
Materials
Lifetime
Mass/Volume Limitations
In-situ Prop. Utilization

LET'S GO TO MARS!

FUSION PROPULSION

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N91-28217

Space Fusion Energy Conversion using a Field Reversed Configuration Reactor

A New Technical Approach for Space Propulsion and Power

PENN STATE SPACE TRANSPORTATION PROPULSION TECHNOLOGY SYMPOSIUM

JUNE 25-29, 1990

Norman R. Schulze, NASA Headquarters
George H. Miley, University of Illinois
John F. Santarius, University of Wisconsin

SPACE FUSION ENERGY CONVERSION USING A FIELD REVERSED CONFIGURATION REACTOR

A NEW TECHNICAL APPROACH FOR SPACE PROPULSION AND POWER

N. R. SCHULZE
G. H. MILEY
J.F. SANTARIUS

THE CONTENTS OF THIS PAPER REFLECT THE OPINION OF THE INDIVIDUAL AUTHORS, NOT NECESSARILY THAT OF THE ORGANIZATIONS TO WHOM THEY REPORT

ABSTRACT

Fusion energy offers many inherent features which would benefit space flight. If the technology had been developed such that fusion energy conversion were available for space use today, fusion energy would be providing increased safety, reduced flight operational costs, and space mission enabling capabilities. The fusion energy conversion design approach, referred to as the Field Reversed Configuration (FRC) -- when burning deuterium and helium-3, offers a new method and concept for space transportation which high energy demanding programs, like the Manned Mars Mission and planetary science outpost missions require. FRC's will increase safety, reduce costs, and enable new missions by providing a high specific power propulsion system from a high performance fusion engine system that can be optimally designed. By using spacecraft powered by FRC's the space program can fulfill High Energy Space Missions (HESM) in a manner not otherwise possible. FRC's can potentially enable the attainment of high payload mass fractions while doing so within shorter flight times. The time has arrived to initiate a space fusion energy conversion program and in particular to demonstrate the FRC potential for space. In addition to the aforementioned advantages, fusion provides an energy option to fission.

INTRODUCTION

This paper articulates the future space mission requirements for high energy missions and the space program's propulsion and electrical power generation plans for meeting those requirements. Current propulsion R & D activities focus on the technology to gain one or two seconds in specific impulse in chemical propulsion systems. Therefore, the concern, in particular, is whether adequate measures are being taken to assure that future space mission propulsion and power needs will be met on a timely basis where the demand is anticipated to be for high energy levels.

Emphasis is placed on the theme that as the low energy space missions are completed, a requirement will develop for a high energy mission capability. That mission capability is not being pursued and could very well be a long time in developing. Yet that does not necessarily have to be the situation. Quantum leaps in the national space transportation infrastructure are possible by developing systems which have high specific power and impulse capabilities. It is the intent of the authors to bring forward some new thinking onto what they perceive as a space propulsion/power crisis that can be anticipated, but which can be circumvented by the use of an alternate energy source.

Energy requirements for space propulsion and electrical power will grow. The accomplishment of high energy missions of the type presented in this paper will not be easy to achieve. This class of high energy missions requires commitment now for the space program to realize timely benefits. The long lead time in bringing this striking new capability forward alerts us to the importance of commencing this challenging research early. Now is the proper time to assume the world leadership role in achieving a high energy capability for space use. The ultimate future for the continuation of advancements in space exploration and space science depends upon the development of that high specific energy capability. Space travel's economics will become severe, possibly to the extent of making high energy missions too costly to perform. That will inhibit our ability to press forward with more ambitious missions. Therefore, the initiation of a relatively modest investment now for a well-planned experimental test program, one designed to achieve a high energy space mission capability, constitutes a major investment opportunity for the future of the U. S. space program and a major challenge to address. This report recommends the use of fusion energy to perform the missions. Its potential offers such great dividends that it can not

be ignored. It is a key element in the implementation of the "US National Space Policy" (ANOM89).

THE POWER OF SPECIFIC POWER

An analysis of future anticipated HESM (High Energy Space Missions), such as the Manned Mars Missions, shows serious shortcomings with the implementation of those missions particularly with regard to the chemical propulsion vehicle's performance, safety, economics, and environmental issues which can become involved with its repeated use for future exploration missions.

Manned Mars settlement, to be successful, will require two high power consumption functions: transportation logistics and local Martian electrical power, subjects which need to be more fully addressed. Since the 1960's, the focus on propulsion systems for Manned Mars Missions has been on chemical propulsion systems combined with aerobraking as the joint technological approach for meeting the mission's energy requirements. Some consideration was given to nuclear fission thermal propulsion, but the performance, operational simplicity, and safety issues detracted from its further consideration. There is recently renewed interest in nuclear thermal systems. Fusion energy has yet to be considered either as a propulsion system option or as an improvement over fission.

Also, there are 2 major applications of a high energy source for electrical power. A large electrical power capability will be important for Mars settlement enabling the utilization of local planetary resources which in turn will reduce the space logistic requirements. High energy levels will provide the space based power for the production of electricity for extraterrestrial settlement including habitat environmental conditioning and manufacturing. The technology to accomplish the utilization of local planetary resources is being pursued by the University of Arizona. It will provide the electrical power for beam power as a potential optional method for providing a cost effective space transportation propulsion logistics support capability. The requirements and methods for high electrical power generation on the planets is not a resolved subject.

The key for the accomplishment of the anticipated high energy space missions, whatever the application, is high specific power. The proper manner by which to address each of the aforementioned issues is through a total space systems engineering approach as discussed in this paper.

Manned space flight safety is achieved by faster trip times resulting in reduced hazard potential from exposure to galactic and solar radiation as well as adverse psychological and physiological effects that could result from long flight times in space. From the perspective of the space traveller, spacecraft having greater mass performance potential will obviously possess the capability to provide more safety features and protection from radiation as well as to provide for other safety features, increased design margin, and back-up flight systems. But from the aspect of safety to the Earth's population, the preference is to place the minimal mass into orbit. Minimal mass also reduces the impact to the environment and the overall economic impact of high energy space missions.

The problem is, how does one resolve these two counterbalancing forces? The solution is to develop high specific power energy conversion systems. High specific power systems, which only fusion energy is currently perceived as capable of delivering, will improve launch safety by minimizing the number of LEO launches. An optimization of mass to LEO also minimizes the energy requirements on Earth's resources that will be necessary to implement the missions. Also minimized are the atmospheric pollutants and the cost of future space flight operations and programs. Nuclear fission propulsion was examined in the 1960's as an option and considered not to be of benefit to the Manned Mars Mission as defined then. There will always be a question, too, of safety from the presence of a large NERVA category power source in Earth orbit and from the ground testing to qualify it. Nuclear electric propulsion does not appear to offer the performance advantage for the large payloads that the Manned Mars Mission requires. It still requires a reactor for power.

The most attractive option is fusion energy. But fusion energy has not been developed to a point where net power has been demonstrated. Even if it had been demonstrated, the experiment which is most likely to demonstrate fusion first is the tokamak. The tokamak is not a concept which can provide the performance necessary to realize the desired advantages. Its large magnet mass prohibits the low flight system mass required for space transportation flight. Instead a light weight concept such as a compact toroid, e.g., the FRC (Field Reversed Configuration), is considered to offer the greatest potential for development.

Properly developed, space fusion energy will revolutionize space travel. For example, if a flight weight propulsion system can be designed having a specific power of 1 kW/kg, the number of Shuttle launches to LEO to perform one Manned Mars

Mission could be reduced by a factor approximately 7 fold from that required by current chemical space propulsion systems. The flight time could be reduced to a total of less than 6 months whereas the chemical propulsion system will require 1 to 2 years total flight duration.

Space program resources must be directed toward those issues as a matter of top priority in undertaking an advanced mission development program. A program designed to test evaluate the FRC reactor burning D-^{3}He could be accomplished on an expedited basis with initial results anticipated within 5 to 10 years.

A High Energy Mission Requirement Exists

This paper first considers hypothesized high energy missions. The energy requirements to meet those missions were analyzed. The results reveal very significant benefits for science and solar system exploration that can be attained by fusion's presence. The practical applications of fusion all relate to large energy consumption missions, namely, those in the multimegawatt category and higher; fusion is not currently foreseen as a competitor to, nor a replacement for, the conventional low energy systems for the near term applications.

The thesis of this report is that (1) a high energy space mission capability needs development (Figure 1a and b) and (2) the Field Reversed Configuration magnetic confinement fusion reactor, burning deuterium-helium-3 is the optimal approach which should be pursued at the highest priority level to meet this need.

THESIS

HIGH ENERGY SPACE MISSION (HESM) CAPABILITY: MULTI MW'S TO GW'S

- Requirement for HESM exists:
 - Manned Mars:
 - Science outposts including sample returns: outer planets, comets, asteroids, others
 - Oort Cloud/Stellar
- Technology lacking. Start R & D now since development will require time.
- Space program's advancement hinges upon high energy conversion elements being made available for the NASA space transportation infrastructure.

Figure 1a. Thesis.

Mission Beneficiaries from High Energy

High Performance Propulsion

- Systematic exploration of Mars, including manned exploration:
 - Safety
 - Economics
 - Reliability
 - Logistics
 - Electrical Power
- Enables: scientific exploration of the entire solar system, interstellar space, and nearest stars.

Figure 1b. Mission benefits from a high energy capability for space.

Improved crew safety results because of the reduced flight time, thereby reducing the crew's exposure to galactic cosmic rays plus other safety factors pertaining to reduced flight times as discussed later. High specific power systems, coupled with a variable

high specific impulse capability, will reduce the launch load requirement over lesser energy intense systems, thereby reducing the quantity of mass which must be placed into low Earth orbit. Many 10's of billions of dollars savings in launch costs can be achieved over low performance systems in implementing a permanent presence program like manned Mars. Reliability gains must be incorporated into remote manned missions, like those to Mars. Reliability will be an ever increasing factor in the accomplishment of future science missions as the flight times become longer, the distances greater, and the mass demands increased for the conduct of more sophisticated missions, such as sample return missions. The brute force method of more redundancy and lower stress through higher safety factors exacerbates the mass-economy problem. New approaches that reduce moving parts and which inherently contain fewer or no parts that are subject to erosion must be incorporated into the flight systems as a new technical approach. A permanent presence of man on Mars will require space logistical support that will be enabled by the space program's capability to support flights there on a frequent basis but which will not be exorbitant in terms of flight costs. To achieve a permanent presence of man on Mars, more emphasis will be placed on self reliance which in turn will necessitate the use of the Martian planetary resources. Significant electrical power will be required to accomplish the manufacturing of the essential products there and to support life habitats. High electrical power technology must, therefore, become a part of the future space mission enabling infrastructure.

Thus, the objective of this paper is to address the concern regarding high energy needs for future space missions and to forward some new thinking on solutions. (Figure 2)

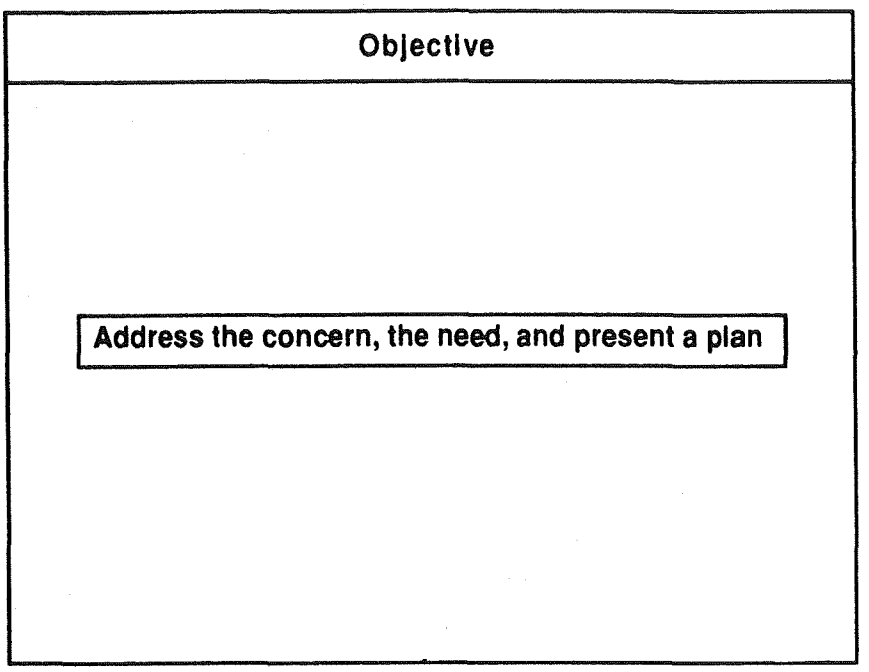


Figure 2. Objective.

The thesis is that a requirement exists for high energy mission capabilities which needs to be addressed. This paper thus examines the missions, system requirements, the basic method to address the system requirements, energy options, and relative advantages; and it recommends a particular energy approach and in particular, a design solution that appears to offer intrinsic advantages which meet space system requirements.

HIGH ENERGY MISSIONS

A few of the high energy missions that can be accomplished if fusion were available include: faster and therefore safer manned Mars missions, manned missions beyond Mars, in-situ stellar science, interstellar plasma science, understanding and mapping of the heliosphere, interstellar astronomy, Oort Cloud exploration and science, multiple planetary outpost missions using just one spacecraft as a launch platform on a single mission, comet/planet rendezvous with sample returns, polar solar science, faster trip times to the outer planets with more massive and better equipped science payloads, science missions to the inner planets, power generation for permanent manned and unmanned science outposts, remote planetary materials processing energy, plus others. Those missions determine certain fundamental system

requirements, Figure 3. The calculations are based upon relatively low thrust, constant acceleration propulsion systems (FRI89).

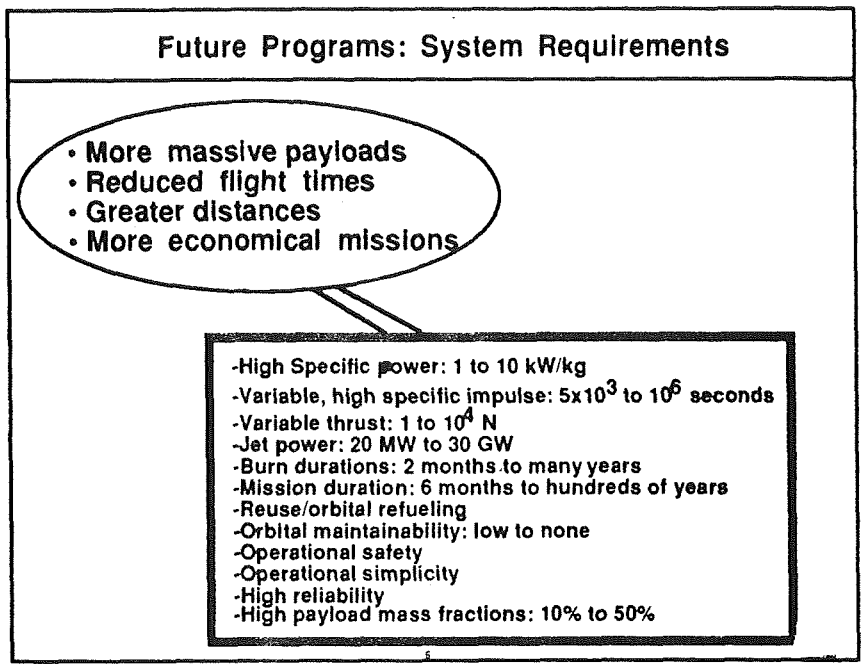


Figure 3. Flight system requirements for future programs.

MANNED MARS EXPLORATION PROGRAM

High specific energy propulsion and power systems particularly benefit the Manned Mars Missions. In addition to Mars, it is anticipated that large power levels will be required for lunar operations to perform mining, material processing, and life support functions. Figure 4a summarizes the key mission design data for future missions, manned and unmanned. A range of values is included showing data for a rapid trip as well as trip times offering economy of propellant and fusion vehicle size while still accomplishing the same mission objectives in a reasonable flight time. The flight time to deliver a 133 MT manned payload to Mars and to return a 61 MT payload to Earth can be accomplished in a trip flight time of 3 months each way with a space launch vehicle of moderate (~610 MT) initial vehicle mass in LEO (Low Earth Orbit) using a propulsion system having a specific power (designated as α_p where α_p = jet power/inert propulsion mass) of 1 kW/kg (Figure 4a and 4b) (FRI88).

Manned Mars Mission Parameters

ADVANTAGE OF HIGH SPECIFIC POWER + HIGH SPECIFIC IMPULSE

High Energy Performance for 133 MT out-bound payload, 61 MT return.

α_p, kW/kg	Flight time, years	M_0, MT	M_p, MT	γ, %	P_j, MW	<isp>, seconds X 10^3	Δv, km/s
1	0.44 - 0.5	1041 - 613	681 - 335	12.8 - 22	227 - 145	9.4 - 10.6	98 - 90
10	0.18 - 0.5	1034 - 185	676 - 30	12.9 - 72	2255 - 227	18.9 - 35.8	196 - 90

α_p: propulsion system specific power, kilowatts/kilogram
Flight time, years: the total round trip time in flight exclusive of visit time at the destination
M_0: initial mass in low Earth orbit
MT: metric tons
M_p: propellant mass in metric. Includes fuels plus diluent
γ: payload mass fraction
P_j: jet power
MW: megawatts
<isp>: average specific impulse over the mission
Δv: incremental velocity change

Reference: Alan Friedlander and Jim McAdams, Letter, dated October 13, 1988, Subject "Completion of Fusion Propulsion Analysis Support," SAIC, Schaumburg, IL

Figure 4a. Manned Mars mission performance using high energy propulsion.

That time could be reduced to a very attractive, short flight time of only approximately one month, provided that a propulsion system having a specific power of 10 kW/kg can be achieved and an initial vehicle mass of ~1,100 MT is placed into low Earth orbit. Refer to Figure 4b for the mission performance characteristic trend curves.

Manned Mars Initial Vehicle Mass

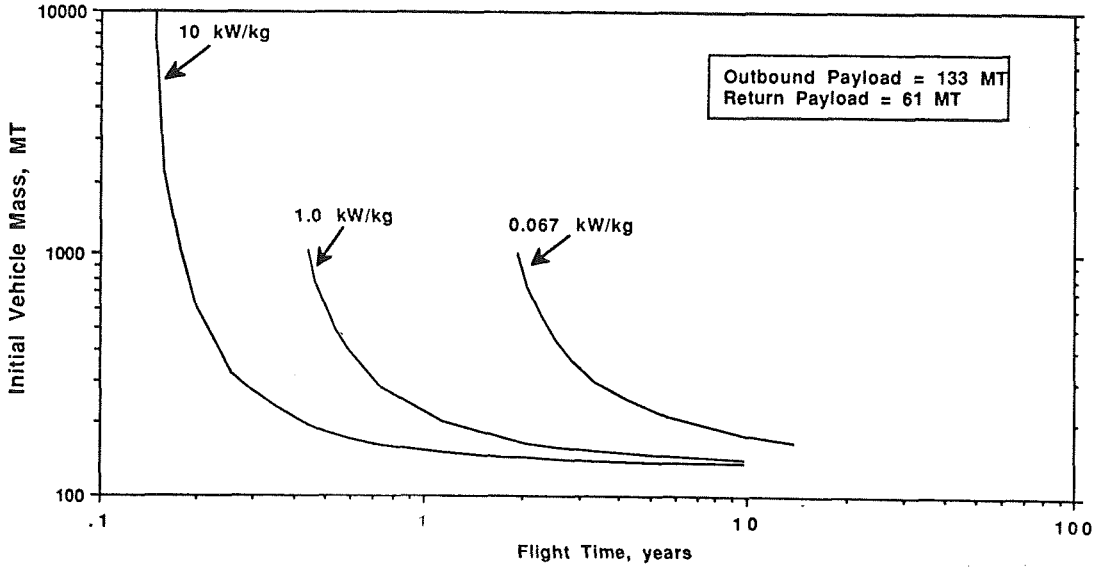


Figure 4b. Manned Mars mission performance using high energy propulsion.

To achieve the anticipated need for more massive payloads, quicker flight times, greater distances traversed, all at reduced costs, the suggested approach is to perform those missions by the development of propulsion systems that yield a high specific power, α_p=1 kW/kg to 10 kW/kg, at a variable high specific impulse (~10^3 - 10^5 seconds). The specific parameters, as specified in Figure 3 will be important to the HESM category.

The economy -- and safety -- goals are attained by substantially increasing the payload mass fraction as shown in Figure 4a. The Shuttle's mass fraction, for example, is low -- slightly greater than 1%. Economy of mission must ultimately be achieved in space as with commercial airlines or other successful transportation businesses, where the payload mass fraction is high. In current day wide body aircraft it is approximately 50%. The mission parameters shown cover a wide mission range, perhaps a full spectrum, of space mission requirements -- from manned Mars, to outer planetary sample returns, out to a rendezvous with Alpha Centauri. The value of high α_p to the manned Mars program is clearly illustrated by Figures 4a and 4b where the flight time, system performances, and masses required to conduct missions carrying 133 MT outbound and 61 MT inbound manned payloads are presented. The α_p of 0.067 kW/kg is considered a target for nuclear electric propulsion. Preliminary studies indicate that fusion can produce specific power

performance in the range of 1 to 10 kW/kg, roughly an order of magnitude above the target for nuclear-electric.

A space logistics infrastructure is basic to the implementation of a viable permanent presence of man on Mars. It is difficult to conceive of a flight frequency less than 2 flights per year. But using current propulsion technology with consideration to its innate performance limitations, the Earth to LEO transportation requirements will be enormous. For example, if launched today using current space propulsion technology, an initial 1,000 MT mass in LEO for the Martian space vehicle would require the energy equivalent of ~37 Shuttle launches. Thus, each flight to Mars, assuming a $320M cost per Shuttle launch, the current cost number, will be at a price of $12B per flight. Larger launch vehicles will obviously reduce the number of launches, but an accurate total systems cost analysis must be accomplished before cost savings can be stated. A specific power of 1 kW/kg propulsion system would permit a 131MT outbound-61MT inbound payload to be sent to Mars using propellants placed into orbit by approximately 6 Shuttle launches, or for a 10 kW/kg system, only one Shuttle launch to deliver propellants to a reusable, space based fusion engine system.

The resolution of high energy space transportation propulsion infrastructure resides not in the capability to launch greater vehicle mass from Earth to LEO and in performing developmental research that yields 1 or 2 seconds improvement in specific impulse. Instead, the space program will better benefit by the development of the technology which requires less mass being placed into LEO to accomplish the same mission, or better still, to accomplish the mission with a more massive payload, flown at higher speeds. That is, a space propulsion system having high specific power and variable high specific impulse is needed.

SCIENCE MISSION PERFORMANCE

While fusion may offer the greatest immediate mission enabling value to the space exploration program, and particularly to the safety of manned missions, fusion energy enables very interesting space science missions. The high energy science missions include soil sample return missions from the moons of the outer planets with **round trip flight times** varying from 1.6 years for Europa to 7.4 years for Charon (Figure 5).

Science Program Benefits

Sample return missions:
20 MT outbound; 10 MT inbound

Mission	t, years (Round trip)	M_o, MT	M_p, MT	γ, %	P_j, MW	<Isp>, seconds x10^3	Δv, km/s
	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$	$\alpha p1 : \alpha p10$
Europa	1.56 : 1.56	320 : 32	243 : 6.8	6.3 : 63	57 : 50	17.7 : 64.1	209 : 209
Titan	2.99 : 2.56	74 : 29	36 : 5.3	27 : 68	18 : 40	26.2 : 81.2	196 : 223
Miranda	5.34 : 5.34	60 : 26	26 : 3.4	33 : 77	14: 27	35.7 : 118	233 : 233
Triton	5.85 : 6.85	108 : 27	62 : 3.8	19 : 74	25 : 30	35.1 : 130	314 : 283
Charon	7.42 : 7.42	81 : 27	41 : 4.1	25 : 73	19 : 32	40.5 : 137	317 : 317

Figure 5. Performance advantage of high specific power/specific impulse propulsion systems for planetary missions (FRI89).

Those times are for the round trip flight time, exclusive of the stay time for science gathering at the site. In the analyzed mission scenarios a very substantial 20 MT payload was flown to the planetary destination and a 10 MT payload returned to Earth where its precious cargo of extraterrestrial soil can be analyzed in depth.

The jet power required to perform such missions is much less than the more massive Manned Mars Missions. It is shown to range from 15 MW to 60 MW. The propulsion system performance is demanding, with the specific impulse ranging between 17,000 seconds and 140,000 seconds. The mission parameters and capabilities for outer planetary missions are summarized in Figure 5. Three separate asteroid visits at 1 AU distance can be quickly performed, i.e., in less than only 2 years using the same 20 MT outbound payload and 10 MT returned payload.

To complete a 10MT payload Oort Cloud rendezvous mission at 20,000 AU, a 700 MW power source operating a 1 kW/kg propulsion system will accomplish that mission in 120 years, while a 10 kW/kg specific power propulsion system completes the trip in 55 years, using a 7 GW reactor power output. The energies here are obviously of a magnitude that a new energy source is mandated. Fusion is a logical

candidate, but serious R & D must begin now in view of the lead time required for such a development.

Our nearest stellar neighbor, Alpha Centauri, actually a 3-star system -- α, β, and Proxima -- at 4.3 light years distance, offers the greatest technical challenge. Alpha closely replicates our sun's characteristics, exhibiting nearly the same brightness properties and mass. But this is not really a mission for a specific power of 1 kW/kg reactor design which takes ~400 years for a 10MT payload fly-by mission, or slightly less, depending upon the initial vehicle mass. For a rendezvous mission, even a specific power system operating at 10 kW/kg requires ~290 years. Advanced system technology might be able to increase the performance capability to 40 kW/kg, thereby reducing the flight time to ~180 years. Because of the mission difficulty it is essential to commence technology development and planning for the mission early.

With fusion, unlike any other known energy source, we can commence consideration of these marvelous missions because of its innate compatibility with high energy mission vehicle system requirements.

VEHICLE SYSTEM REQUIREMENTS

The ability to perform the complete class of missions considered herein resides upon several key factors which serve as the basic high energy system mission architecture requirements, Table 1, for the next generation spacecraft which the United States space program should now be pursuing to assure a national space posture in the future.

Table 1. Future spacecraft energy system needs (SCH90).

1. the ability to develop a specific power system of 1 kW/kg, or 10 kW/kg in the case of the stellar mission;
2. the ability to produce sufficiently high thrust for a vehicle of this size and a variable specific impulse (10^4 to 10^6 seconds);
3. reliable propulsion and vehicle performance for months to many years (e.g., for as long as 50 years of continuous firing operation);
4. reactors ranging from 20 MW to 30,000 MW jet power production;
5. the ability to perform the missions safely from both the standpoint of public safety and flight safety.

The most severe requirements are established by the stellar mission. An orderly progressive reactor enhancement program build-up will ultimately allow NASA to proceed from the lesser demanding missions to the more difficult, for example, the 10's MW for unmanned science payloads to 100's MW for manned missions to GW's for stellar missions. The capability to meet the high energy mission specific impulse and thrust requirements imposed by the vehicle on the propulsion system are to be similarly developed.

PROPULSION SYSTEM REQUIREMENTS FOR HIGH ENERGY MISSIONS

From the mission and vehicle requirements we can determine the fundamental space vehicle propulsion system requirements for high energy missions. These are shown in Table 2 below (SCH90).

Table 2. Propulsion system requirements for high energy missions (SCH90).

- minimize propulsion system mass,
- meet long system life time requirements of years,
- provide a remote, reliable, and efficient space restart capability,
- use only radiation for cooling,
- be designed for the presence of a "free" continuous vacuum,
- provide power for variable propulsive thrust and specific impulse requirements,
- provide sufficient power also for the generation of electricity,
- operate in a low acceleration environment (low thrust and zero gravity),
- produce a very wide range of output power levels (throttable),
- be designed for long operational times - thrusting and quiescent despite a lack of ready access for maintenance.

Space propulsion system requirements can only be met by an effective space fusion research program, one which is conducted on a program priority reflecting the importance of fusion energy to the space mission architecture.

SPACE ENERGY OPTIONS

The available energy options for HESM and specific energy for each are compared in Figure 6. The greater than 7 orders of magnitude improvement in specific energy over chemical is the initial rationale for interest in fusion. The potential for high efficiency energy conversion and other properties, including safety, as discussed

subsequently, make fusion a more desirable energy source for space propulsion than fission, the other high specific energy source shown. The authors considered matter-antimatter as another potential option but have serious reservations concerning its competitiveness with fusion and fission based upon the relative technology data bases at this time. Solar energy cannot serve as a high energy source that will meet the demands of the mission class considered herein.

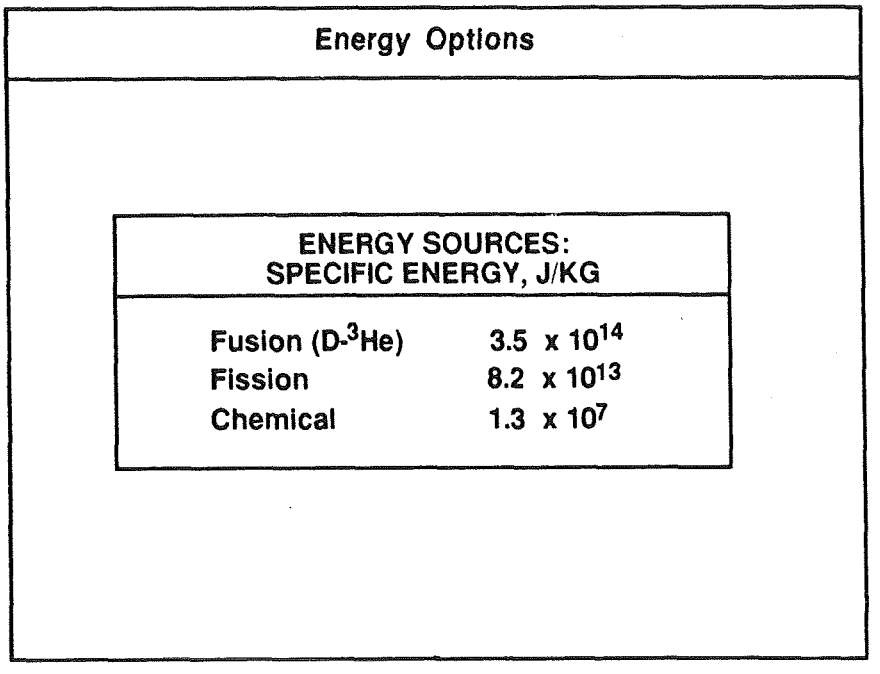


Figure 6. Specific energy for space energy options.

A comparison of the relative merits of the three energy sources and their estimated capability to meet those mission requirements presented earlier are shown in Figure 7. Except for the chemical systems these are subjective evaluations due largely to the undeveloped status of the nuclear energy systems for space.

Preliminary analyses and/or educated guesses. All require thorough analysis, design, and testing to validate whether the parameters can be met.

Desired Parameters and Values	Fusion	Fission	Chemical
High Specific power: 1 to 10 kW/kg	✔	<1	✔
Variable, high specific impulse: $5x10^3$ to 10^6 seconds	✔	?	
Variable thrust: 1 to 10^4 N	✔	✔	✔
Jet power: 50 MW to 10 GW	✔	?	
Burn durations: 2 months to 50+ years	✔		
Mission duration: 6 months to 5 years for solar system missions	✔	✔	✔
Reuse	✔	?	✔
Low to no space maintainability:	?		✔
Operational safety	1	3	2
Operational simplicity	✔		✔
Cost effectiveness for high energy missions	✔		
High payload mass fractions: 10% to 50%	✔		

Figure 7. Comparisons of energy options.

SAFETY

Safety in the figure is ranked highest on a scale of 1 to 3 for the use of fusion, based upon the attributes listed in Figure 8.

Attributes:

Safety is a major motivation for the use of fusion for HESM.

- Faster trips to Mars (~3 months one way).
- Decreases substantially the numbers of launches to LEO
- Propulsion braking, not aerodynamic braking on Mars mission.
- Non radioactive fuels.
- Absence of high speed components such as SSME turbines.
- Fuels do not chemically react with each other.
- Total energy content of plasma is very small.
- Absence of environmental impact on the Earth.

Issues:

- Activated materials from neutrons: resolve by minimizing neutrons + shielding
- Cryogenic fuel storage and magnetic cooling: resolve by standard design and safety practices

Figure 8. Safety implications concerning the use of fusion energy.

Faster fight times minimize the hazards to the flight crew that occur from galactic radiation (reduced integrated dosage) and solar events (probabilistic occurrence), psychological effects from an extended time in a small confined space (without escape), and physiological deterioration from extended weightlessness periods. While all of these issues may have "workarounds," fusion offers significant advantages for reducing the concerns very substantially.

Where high energies are required in space, high specific power systems reduce the mass requirements and consequently the required number of launches to place the mass there. It is obviously safer to place the mass necessary for a Manned Mars Mission into LEO using 5 Shuttle launch equivalent flights rather than 37, for example.

Note that the high level of propulsion system performance permits the use of propulsive, not aerodynamic, energy transfer for braking maneuvers. That provides more flight operational options and greater tolerance to errors and is, therefore, considered as an inherently safer flight operational mode.

Although the neutron flux from the burning of fusion fuels is not anticipated at this time to be entirely eliminated, with the proper selection of fuels it can be reduced to the low value of approximately 1-2%. That aids the design process substantially but is still sufficiently high to activate structural materials and to require some shielding. Most importantly, however, is the avoidance of high level radioactive fission products.

It is important for the next programs to assure safety to ground handing personnel and to the public by the selection of fuels that eliminate radioactive elements. Public opposition concerning these matters is also eliminated.

Magnetic fields provide a very reliable and effective means of confining the fusion plasma and holding it where desired. Magnetic field lines direct the thrust particles. Wear and high kinetic energy components typically associated with conventional propulsion systems are therefore eliminated. For example, nozzle erosion and attendant hazards, as experienced with solid propellant motors, are eliminated as are those associated with high speed turbopumps.

The total energy content of the working "fluid," i.e., the plasma, is small at 10^{15} ions/cc. The primary hazard is termination of the reaction if the plasma should come into contact with the first wall. Damage to the reactor magnet is the worse case. The reactor is not going to "blow-up," in contrast to liquid and solid propellant systems which can occur when internal system divergences are experienced.

Deuterium can be extracted from sea water using solar energy if necessary, and ^3He can be mined on the moon. An option for obtaining ^3He is to breed it on each using a special accelerator-target facility.

The two primary fusion reaction hazards are the presence of neutrons and the use of cryogenic fluids. Other secondary hazards include stored energy in the magnetic fields and high voltages. The proper selection of fuels which minimize the neutron flux, combined with shielding, is the proper resolution of the neutron hazard. The other hazards are controlled by standard, well developed practices for working with cryogenics, static loads, and high fields/voltages.

Let us address the subject of fusion energy and propulsion and the means by which the authors suggest its advantages can be realized.

FUSION REACTIONS

In fusion reactions, under the right set of conditions, light weight nucleons join to form other nucleons; the products are referred to as fusion "ash." Some of the ash is burned in secondary reactions although this is usually a small contributor to the total fusion power. The conversion of mass to a specific quantity of energy is determined by the mass loss between the initial reacting mass and the residual rest mass of the reaction products in accordance with the equation, $E = mc^2$. The energy appears as kinetic energy of charged particles and/or neutrons depending upon the fuels selected for the reaction. The challenge in achieving controlled fusion has been in designing a satisfactory stable confinement scheme capable of containing the high temperature plasma (10^8-10^9 °K) sufficiently long that a net positive yield of energy results. The status now is that we have currently come to a point where the fusion energy production is very close to breakeven, only being down a factor of 3-5.

SPACE FUEL OF PREFERENCE

Of foremost importance is the selection of a proper fusion fuel pair for space use. The number of nature's elements which will fuse is indeed quite large. However, during the discussions on space energy fusion fuel applications we shall be concerned primarily with just three reactions, i.e., those listed in Figure 9a and 9b (group A).

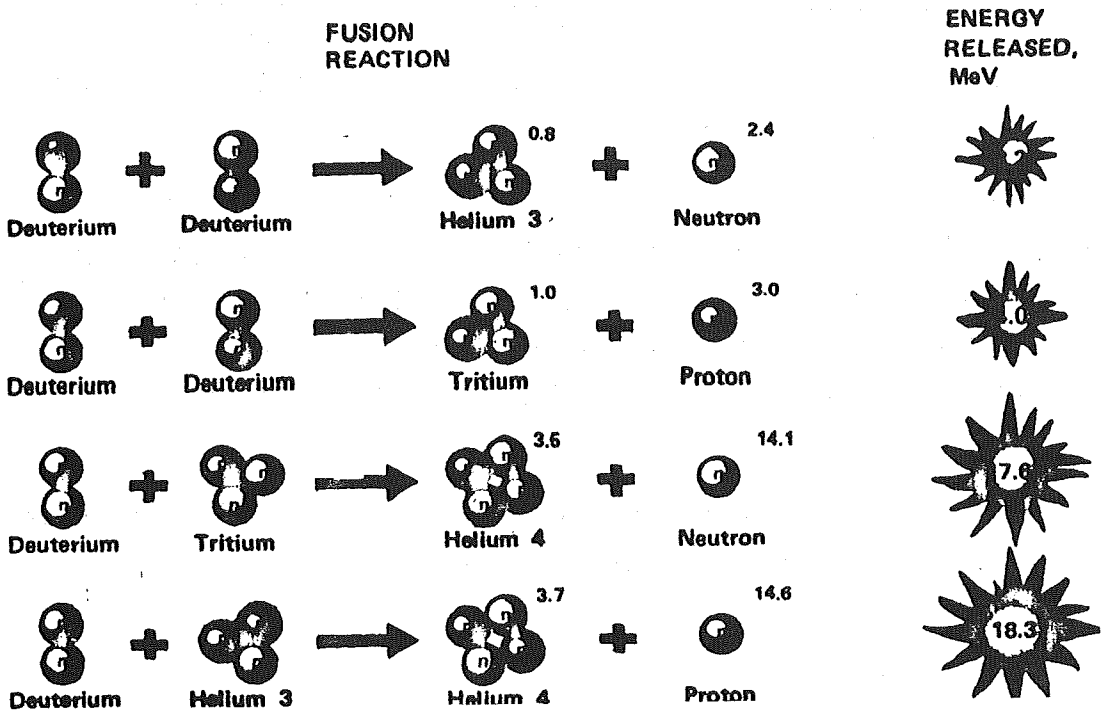


Figure 9a. Fusion fuels for space applications.

Fusion Reactions for Space Applications

A. The most important fusion reactions for space applications

1. $D + {}^3He$ = p (14.68 MeV) + ^4He (3.67 MeV) nearly aneutronic: D-D side reaction)
2. $D + D$ = n (2.45 MeV) + ^3He (0.82 MeV) (50%)
 = p (3.02 MeV) + T (1.01 MeV) (50%)
3. $D + T$ = n (14.07 MeV) + ^4He (3.52 MeV)

B. Other Desired (Aneutronic) Reactions (energetically very difficult)

4. $p + {}^{11}B$ = $3\,{}^4He$ (8.7 MeV total)
5. $^3He + {}^3He$ = 2p (5.7 MeV each) + ^4He (1.4 MeV)

Figure 9b. Fusion fuels for space applications.

Those listed in group B as purely aneutronic, i.e., without neutrons in the reaction products, are preferred; but these reactions are energetically very difficult to achieve,

i.e., a high energy level is required to initiate the reaction to produce net power from the reacting elements. The net power gain is, therefore, very low by comparison.

As shown by Figure 10 the preferred fuel for space is deuterium-helium-3 where nearly all of the energy is present in the form of charged particles, 14.68 MeV protons and 3.67 MeV alpha particles. An assessment of advanced fusion energy for space applications, conducted by the Air Force Studies Board for the National Research Council, reached similar conclusions (MIL87). The confinement conditions required to burn it are less than an order of magnitude greater than the D-T reaction (and much less demanding than the other aneutronic reactions).

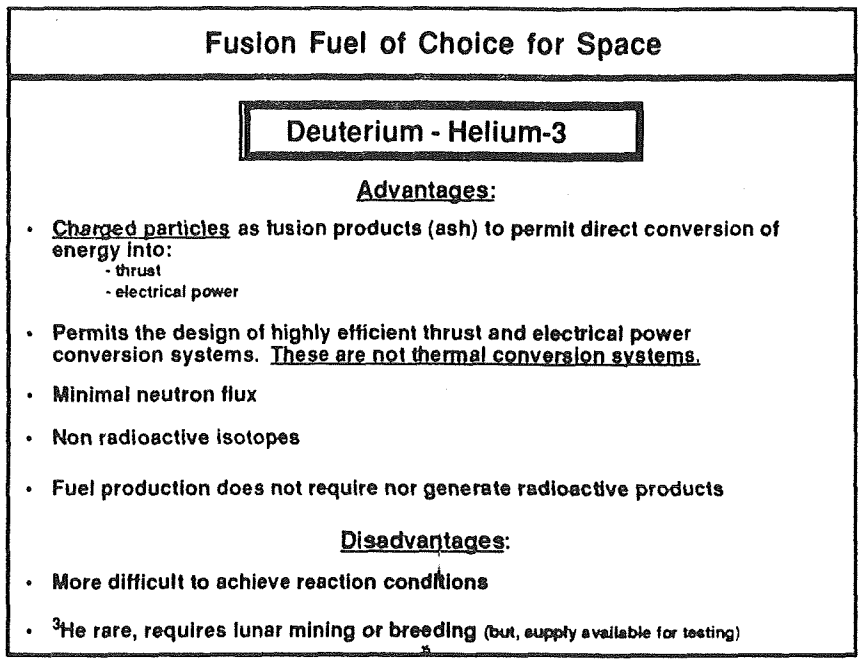


Figure 10. Space fusion fuel preference.

The D-^{3}He fuel cycle is particularly attractive and is preferred over other high energy sources since the charged particles can readily produce thrust by being propelled as magnetically controlled bleed off particles from the plasma through a magnetic nozzle. Note also that high specific power is made possible due to high β (i.e., the ratio of plasma pressure to magnetic pressure = 90%) and the replacement of heavy coils by plasma currents. That important parameter is, thus, made possible by a reactor capable of burning fuels whose reaction products are charged particles. Fortuitously, more than 95% of the D-^{3}He reaction's energy is present in the form of charged particles, namely, alpha particles and protons, the energy of which can be

converted directly to propulsion and/or electrical power without the usual thermal and mass inefficiencies and losses associated with those systems. By the proper use of design parameters the neutron flux can be reduced to approximately 1-2% (CHA89). With regard to its availability, helium-3 can be mined on the moon and has been estimated to contain ~10^9 kg (WIT86), Similarly, it can be expected to be present on other airless bodies. It can be bred using proton acceleration onto lithium-6 or alternatively via the production and decay of tritium (MIL88). There is sufficient helium-3 available now on Earth for accomplishing a meaningful test program without lunar mining preceding a fusion program (KUL87).

To fuse nucleons, several conditions must be met. Sufficient kinetic energy must be imparted to the ions to overcome the mutually repulsive Coulomb forces and to penetrate their respective nuclei. Hence, a large quantity of energy is required to initiate fusion reactions. Whether or not two nuclei fuse is a statistical matter of nucleons colliding at the proper point of impact and with a sufficiently high energy (velocity) to result in nucleon penetration. The rate of reaction (Figure 11) is expressed by $\langle\sigma v\rangle$ which is the average product of the fusion reaction's nuclear cross section area (σ), $cm,^2$ and the relative ion velocity (v), cm/sec. It is referred to as the reaction rate coefficient. The product of the reaction rate coefficient with the energy per reaction determines the energy density.

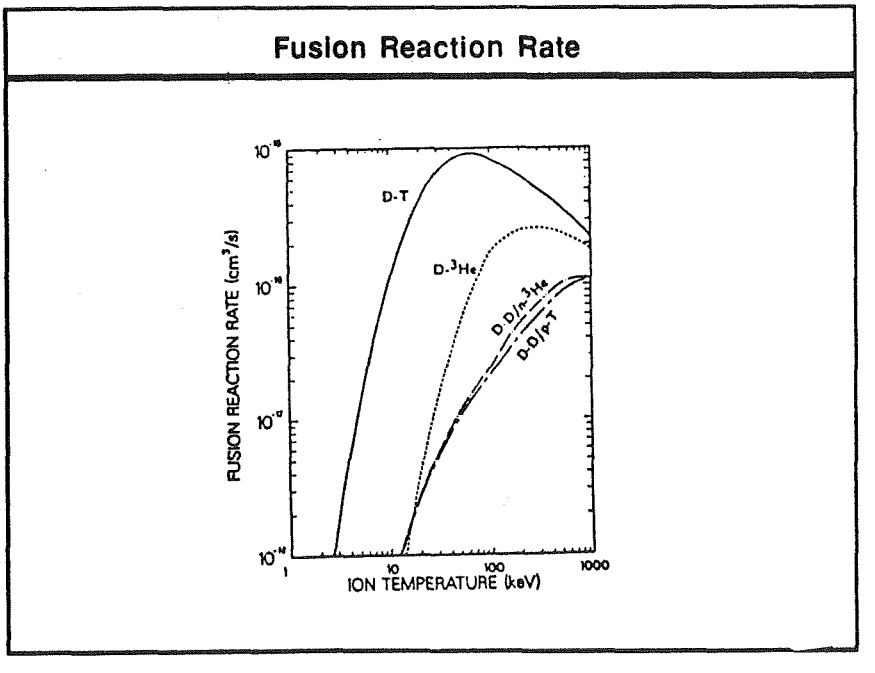


Figure 11. Fusion rate of reaction for selected fuels (SAN88).

The plasma, must be confined for an adequate time (τ), seconds, at a sufficiently high ion density (n), number of ions/cm^3, and at a sufficiently high temperature, T_i, to achieve burning. The confinement figure of merit of a plasma is measured by the confinement parameter $n\tau$ and temperature T_i, Figure 12.

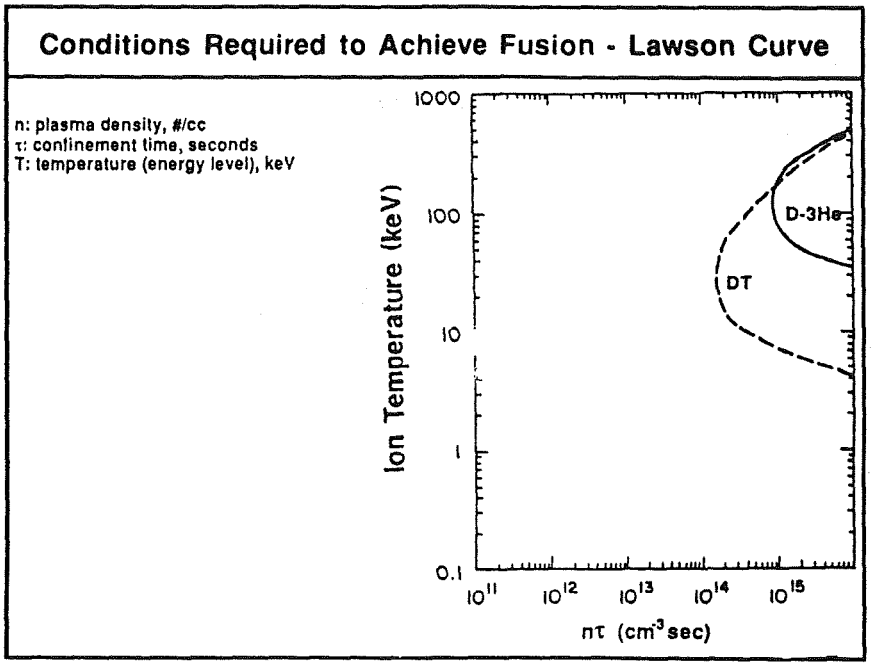


Figure 12. Lawson curve.

Figure 12 presents the Lawson criteria. The Lawson criteria defines the breakeven condition value of $n\tau$ required at a given temperature T_i. Breakeven is the point at which the total fusion output, if it were converted to electricity and reinjected, the reactor would self-sustain burning. This provides an excellent first estimate of these parameters, although Lawson made certain assumptions such as 33% energy conversion efficiency and 100% efficient heating of the plasma by fusion products.

Neutrons, as typical reaction products, are immediately lost from the plasma without a transfer of energy to the plasma. The charged fusion products, i.e., ions, are slowed by the background plasma, and their energy then serves to heat the plasma and any cold fuel input. When the product of fuel confinement time and fuel density ($n\tau$ product) is sufficiently large ($n\tau \geq 5 \times 10^{14}$ cm^{-3}sec where $T_i = 10$ keV for DT and for D-^{3}He, $n\tau \geq 2 \times 10^{15}$ cm^{-3} sec where $T_i = 30$ keV, for example), the charged fusion product heating can balance plasma energy losses from conduction, convection, and radiation as bremsstrahlung and synchrotron radiation. When this condition occurs,

the plasma is said to be ignited, and the burn can proceed without further input of energy from external auxiliary heating systems. The progress made over the past 25 years, Figure 13, shows an improvement of 7 orders of magnitude in the E_{out}/E_{in}, the value of which is rapidly converging on breakeven for the tokamak, the leader in the magnetic confinement experiments.

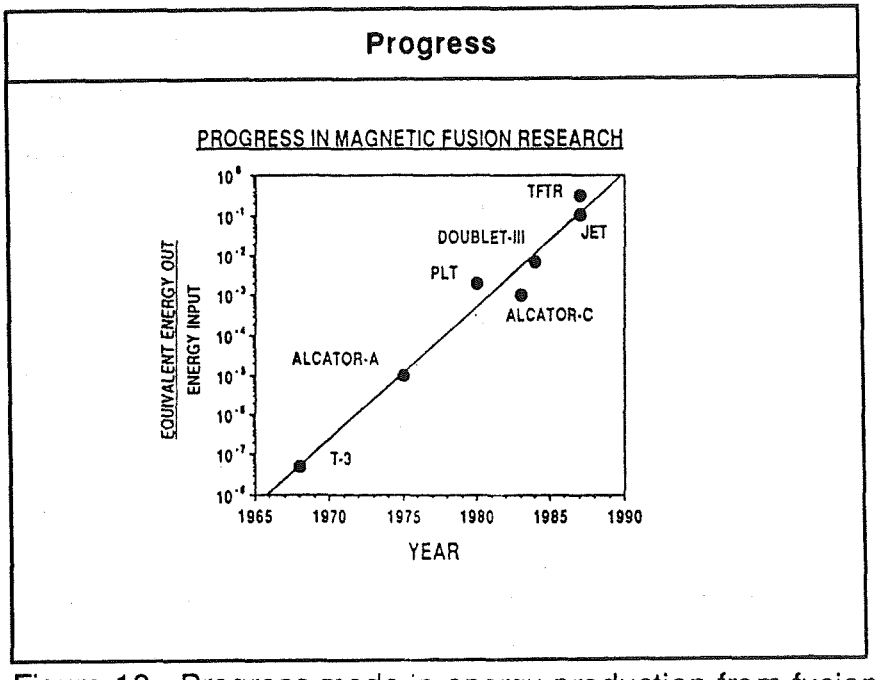


Figure 13. Progress made in energy production from fusion experiments (SAN88).

The status of several key experiments is shown later in Figure 25. The operational regimes for $n\tau$ and T have both been met individually by different experiments, although not at a level that satisfies both parameters, $n\tau$ and T_i, simultaneously.

MEANS OF ACCOMPLISHMENT

There are three ways by which fusion can occur: magnetically confined plasmas, inertially confined plasmas, and gravitational (Figure 14).

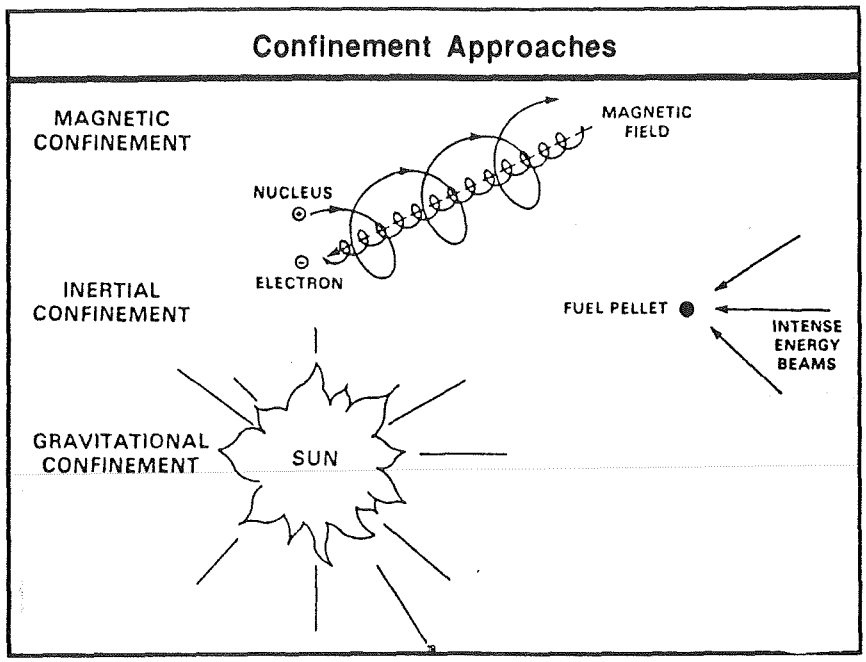


Figure 14. Means to achieve fusion.

Magnetic confinement, the focus of this report, has been researched the longest. The inertial confinement approach uses very high energy laser beams targeted at a small (~1 mm) pellet of fusionable fuels to reach the Lawson parameters under high densities for short periods of time. Efforts at demonstrating a cold fusion process (not presented on the figure) are under study or are uncertain, except for muon catalysis which is not a space option without a light weight accelerator. Figure 15 shows two magnetic confinement approaches, a simple magnetic mirror -- an open system -- and a simple torus -- a closed system. Plasma confinement is provided by magnetic force fields from magnet coil windings. The reactor suggested by this report, discussed next, uses principles pertaining to both, but without the extensive coil windings.

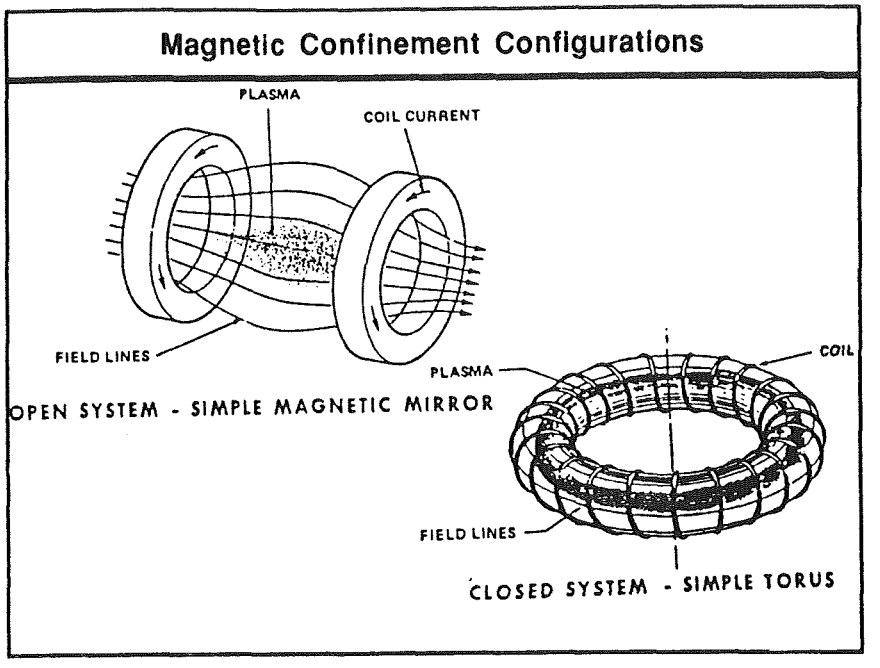


Figure 15. Basic magnetic confinement techniques.

FIELD REVERSED CONFIGURATION (FRC)

When considering the options for magnetic confinement for space we need to evaluate the capability of reactor design approaches that most closely meet space requirements, Table 3.

Table 3. Fusion Options and Comparative Evaluations (CHA89).

Parameter	Field Reversed	Tandem Mirror	Spherical Torus
Specific Impulse	○	○	○
Thrust (Power)	◒	○	○
Beta	○	◒	●
Power Density	○	◒	●
Thrust (Power)/Weight	○	◒	◒
Charged Particle Extraction	○	○	◒
Propellant Thermalization	○	◒	●

○ - Good ◒ - Average ● - Poor

Table 3 shows the Field Reversed Configuration (FRC), of the current magnetic reactor concepts considered applicable to space, to offer the optimal plasma confinement concept (Figure 16), hence the proposed approach of the authors.

Field Reversed Configuration
This paper discusses the design and operating principles of the magnetic confinement reactor known as the Field Reversed Configuration. Its applicability to the space program is examined and shown to be potentially beneficial. A FRC developmental plan is outlined.

Figure 16. FRC content.

The FRC's characteristic plasma ion flux is illustrated in Figure 17 by the arrows in the torus.

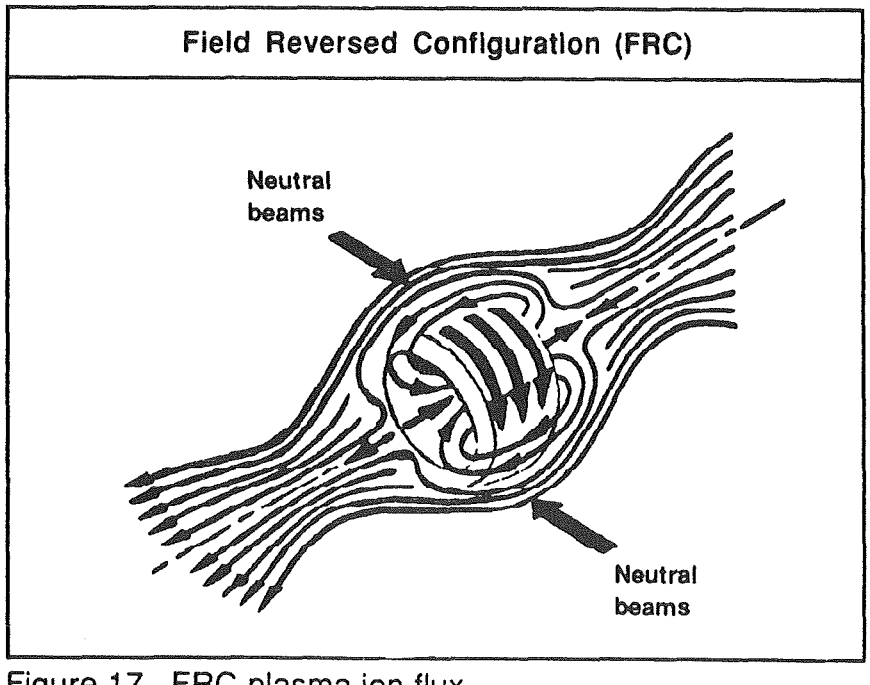


Figure 17. FRC plasma ion flux.

The FRC combines attractive features of both toroidal and linear systems. The closed inner field surfaces provide good confinement of the plasma. Yet, the linear topological nature of the external magnetic field lines would be conducive to the production of direct thrust.

The attractiveness of this machine stems from its high β good plasma confinement scheme, high power density, potential for steady state operation, and overall compact design. Plasma confinement is provided by the two end magnets and a reversed field which may be initiated and sustained by a number of methods. A toroidal current produces the confining magnetic lines of force which are in the poloidal direction (refer to Figure 18).

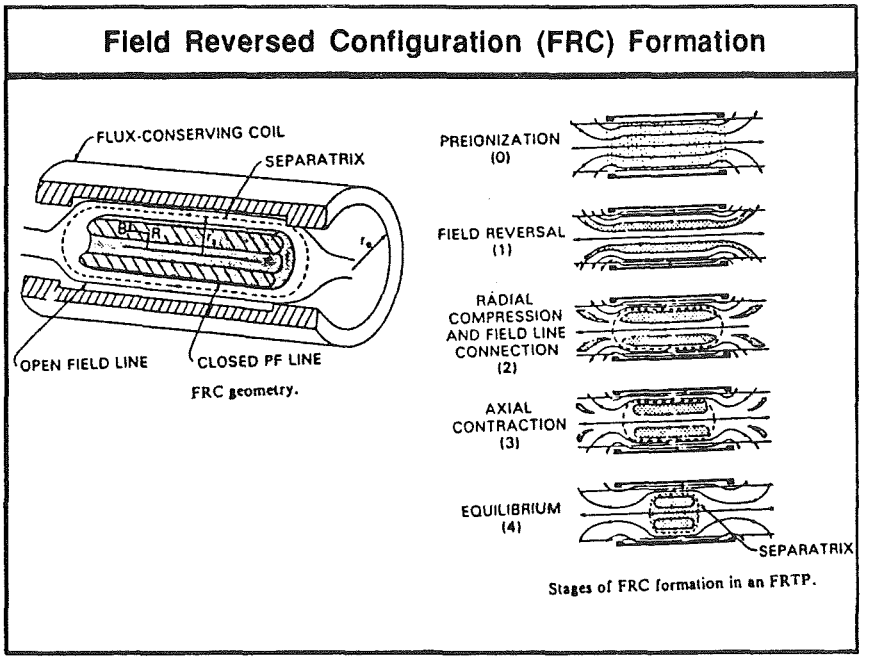


Figure 18. Plasma formation in an FRC (HOF86).

The FRC's advantage resides with the device's innate ability to contain the fusion plasma with a magnetic field generated by large internal currents that are produced without requiring magnetic coils linking the plasma. The plasma formation steps are shown in Figure 18.

One possibility for achieving ignition is to heat the fuel to the ignition temperature by quickly compressing the plasma with a rapid ramping of the plasma current and an increased magnetic field. Another is to inject a high energy neutral beam. The

plasma fusion products heat the surrounding plasma, providing an attractive reactor energy balance.

The optimism for the FRC's performance as a viable reactor is indicated by the statement made by Dr. Tuszewski, one of the FRC scientists at the Los Alamos National Laboratory, in a paper presented at the Eighth Topical Fusion Meeting (TUS88). "The FRC is ideal for use of the D-^{3}He fuel cycle. Its high plasma beta and power density allow substantial reactivity, little radiation losses, and most of the fusion power in the form of 14.7 MeV protons. These charged particles can be diverted in the FRC edge layer towards electrostatic direct converters, resulting in very high plant efficiencies. These attractive features are illustrated in Table 2, where the approximate parameters of a 1 GW FRC reactor are compared for a pulsed D-T system such as CTOR and for a conceptual steady-state D-^{3}He system. One observes that the 14 MeV neutron production with D-^{3}He can be reduced by about a factor 100 compared to that of the D-T system. Another (possibly crucial) advantage of the D-^{3}He system is that gross FRC stability may be achieved at $s \sim 10$ with the help of high energy neutral beams, large-orbit protons, and possibly larger plasma elongations. This may not be the case for the D-T pulsed system at $s \sim 30$, in spite of the alpha particles."

Two terrestrial FRC experiments are in operation, one at Los Alamos and another at Spectra Technology in conjunction with the University of Washington. The FRC's fundamental advantages are presented in Figure 19.

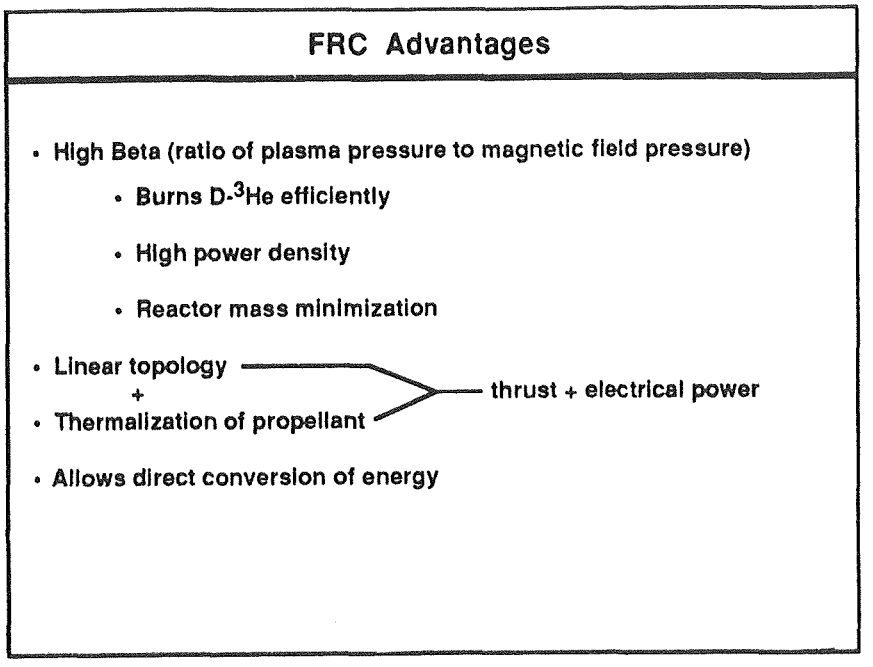


Figure 19. Inherent advantages of the FRC plasma confinement design.

The capability of the FRC to meet the space requirements as defined by Figure 3 is considered to be a good match. Thus, it appears to have very desirable inherent properties for the space application -- Figure 20.

FRC Status: Space Requirements Compatibility

Preliminary analyses and/or educated guesses. All require thorough analysis, design, and testing to validate whether the parameters can be met.

SPACE REACTOR PARAMETER GOAL	FRC PERFORMANCE AND RESEARCH STATUS			
	Meets	Does not meet	Unknown	Comments on FRC status
Specific Power Thrust	✔			Limited study. Requires design.
(a) low: 1 N to 10K N	✔			Limited conceptual work.
(b) medium: 10KN to 50KN			✔	Has not been addressed.
(c) high: 50KN to 500KN		?		Requires a large plasma volume.
Specific Impulse	✔			10^3 to 10^6.
Fuel Cycle			✔	Can burn D-^{3}He
Beta	✔			90%.
Ignition	✔			Needs design study. Requires testing.
Throttle capability	✔			Needs design study. Requires testing.
Plasma Stability			✔	Limited analysis. Burn experiments required - major issue.
Power Level	✔			~50 to 10 GW
Electrical Power Variability	✔			Requires design study.
Dual Mode Operation	✔			Requires design study.
Mass			✔	Requires design study.
Efficiency (P_{thrust}/P_{fusion})	✔			High.
Recirculation Power	✔			Low.
Modes of Operation	✔			Work will follow net power demonstration.
No (low) Neutron production	✔			<2%.
Failure Tolerance	✔			Requires design study.
Space Environment	✔			Requires design study.

Figure 20. Compatibility of the FRC with space reactor design requirements (SCH90).

The evaluation must necessarily be considered as subjective due to the lack of any study or testing which will support the conclusions with data. Note that the key parameters, such as plasma stability, require further investigation, the basis for establishing a space fusion propulsion developmental plan.

FUSION ENGINE DESIGN

The FRC is ideally suited to propulsion by virtue of its external topology. Engine thrust is produced by the controlled release of a portion of the plasma, directed by a magnetic nozzle. One advantage of magnetic reactor designs is the absence of moving parts and of parts subjected to erosive wear. These are essential, inherent features to achieve the long life time operational requirements of the space program. The reactor is fueled by pellets which are injected into the plasma. Thrust and specific impulse are simultaneously controlled by the injection of propellant into the scrape-off layer. The thermalization of propellant is attained by heating from the plasma; the extent of thermalization is important to assure its efficient use. Plasma thrust is produced and controlled by the release of plasma and propellant along the axis through the external mirror magnets. A reactor of the power magnitude required by the manned programs would be characterized by the parameters as shown by Table 4 below (CHA89).

Table 4. FRC High Power Design Parameters.

Total power	0.5 GW
Plasma Volume	80 m^3
Elongation Factor	6
Ion Gyro Radius	0.01 m
Plasma Radius	1.5 m
Stability Factor	50
Propellant Addition	0 - 0.8 kg/s
Specific Impulse	10^6 - 10^3 seconds
Thrust	0.4-50 kN

Thrust for a fusion engine is produced directly by a magnetic nozzle at one end, accomplished by a field imbalance, Figure 21. The thrust and specific impulse are varied by changes in the propellant flow rate.

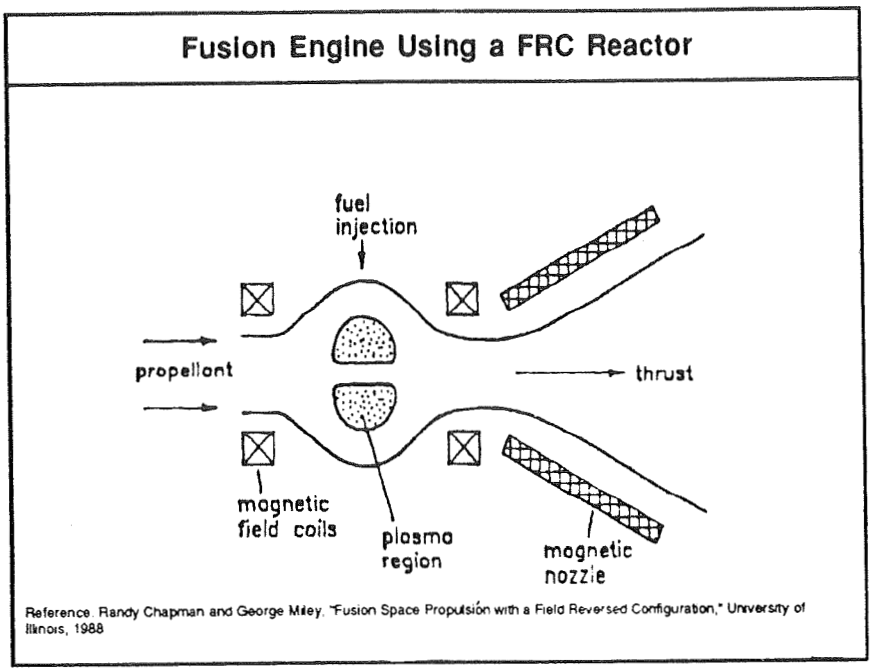


Figure 21. Fusion engine design concept (CHA89).

Fusion propulsion performance is shown by Figure 22 for three operational modes: the highest, plasma only at 10^6 seconds; a variable range attained by the injection of a diluent; and a thermal conversion mode comparable to any thermal propulsion system. Thrust is increased as specific impulse decreases.

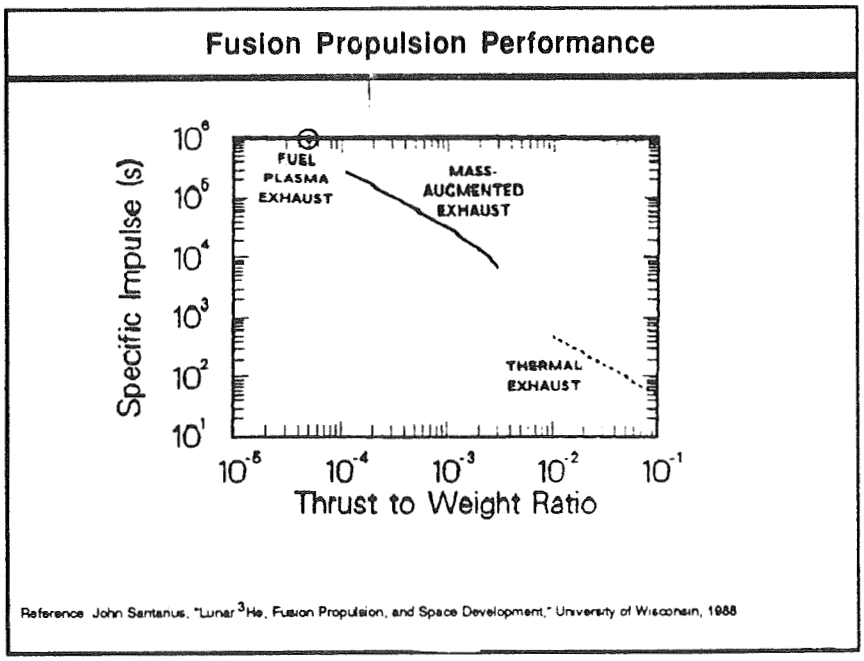


Figure 22. Fusion engine specific impulse performance (SAN89).

The use of the magnetic nozzle and plasma entrapment makes this concept attractive because the plasma remains physically away from the wall.

TECHNICAL CONCERNS

The concerns that need to be addressed are shown in Figure 23.

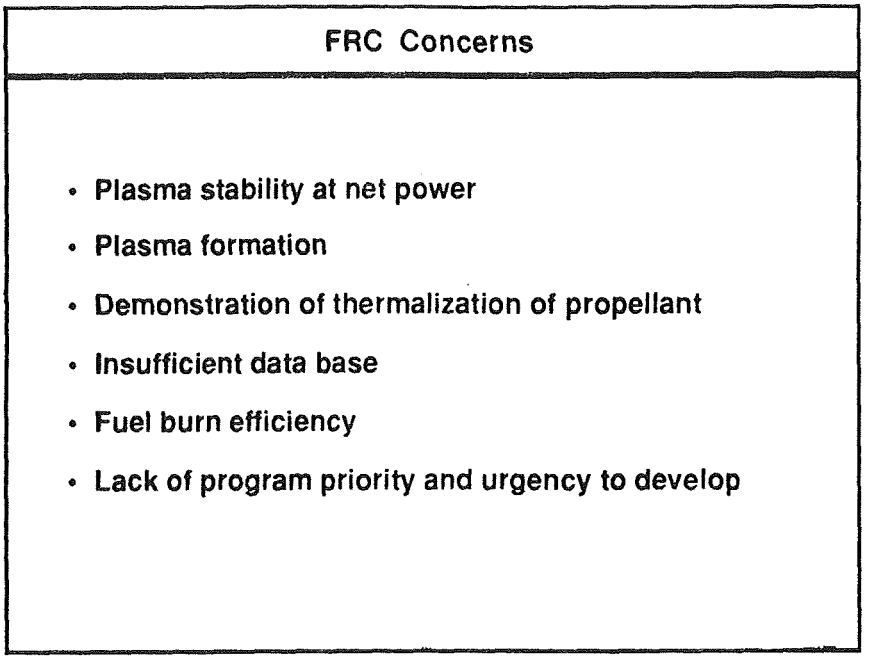


Figure 23. FRC parameters requiring further and testing.

The FRC's limitations that need to be addressed are as follows (Table 5) (CHA89, SCH90):

Table 5. FRC limitations.

- Limited volume: Its size is considered to be volume limited based upon stability considerations. One approach taken to produce greater power is to provide a greater elongation factor. This consideration may be the ultimate limitation on the reactor size. Ions injected to orbit the plasma are anticipated to assist in the maintenance of plasma stability.
- Fuel efficiency: One important subject for investigation is the means to improve upon the fuel burn-up factor which is ~3%.
- Reactor plasma efficiency: Thermalization efficiency of the propellants, ash, and reaction products must be studied in detail.

Much of the concerns result from the fact that relatively little emphasis has been placed on the FRC. Consider the status as shown in Figure 24 which shows that the FRC resides in the least developed knowledge base.

Reactor Knowledge Base

Magnetic Confinement Concept Classification of Reactor Knowledge Base.

Well Developed	Moderately Developed	Less Developed
Tokamak	Advanced Tokamak Tandem Mirror Stellarator Reversed Field Pinch	Field Reversed Configuration Spheromak Elmo Bumpy Square Dense Z Pinch

Figure 24. Comparison of fusion experiment knowledge base (SCH90).

When we consider its demonstrated $n\tau T$ performance relative to ignition for other reactor experiments, the advancement is not nearly as great, largely due to the few FRC experiments built to date. (Figure 25) That chart, in essence, summarizes the FRC development risk.

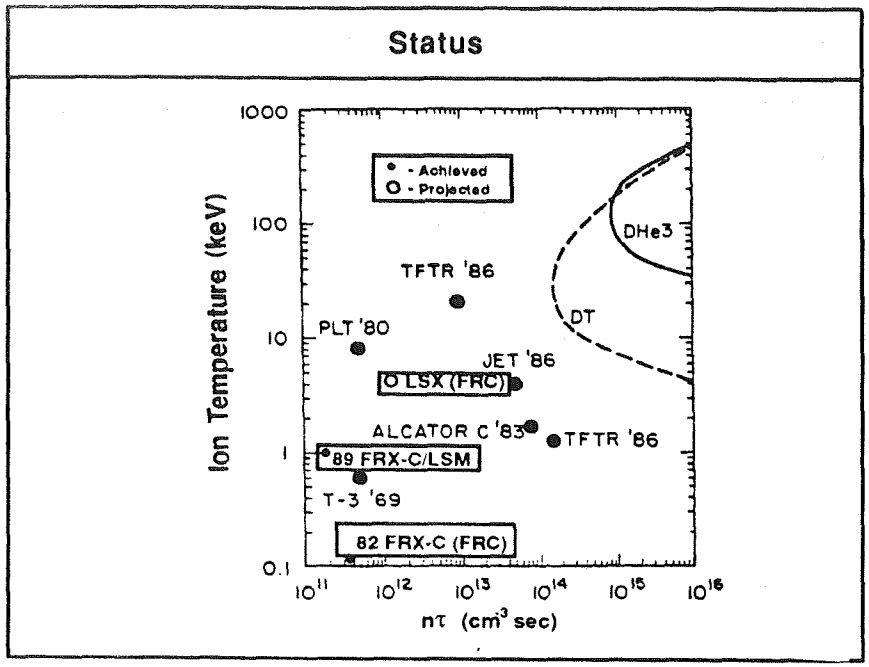


Figure 25. Status of fusion experiments relative to meeting burning conditions (SAN88).

Space Vehicle System Issues

Simultaneously with the development of the capability to produce thrust from controlled fusion is the ability to provide technology for the system capabilities that will satisfy the mass constraints necessary to achieve the specific power for these systems. Refer to Figure 26.

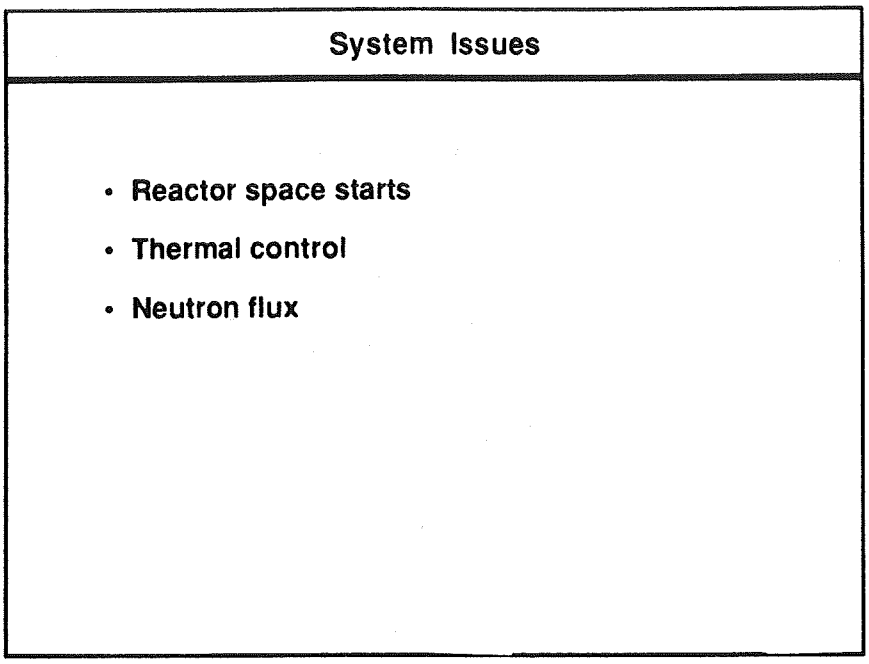


Figure 26. Significant space flight fusion system issues that need to be addressed (SCH90).

The means to provide an in-space restart capability within specific power constraints constitutes fundamental supporting space fusion technology research. Yet no such research effort is being expended. Thermal control and neutron flux abatement are the other two key technology issues to make fusion energy practical. The selection of D-^{3}He as the space fuel is important in order to simplify the system engineering task and to minimize mass. The space restart technology is the most key topic in need of R&D consideration since large levels of energy will be stored aboard the spacecraft to restart the reactor. The production of highly effective, low mass, electrical power systems for space applications needs to be further researched.

Costs

The status for program costing is shown in Figure 27.

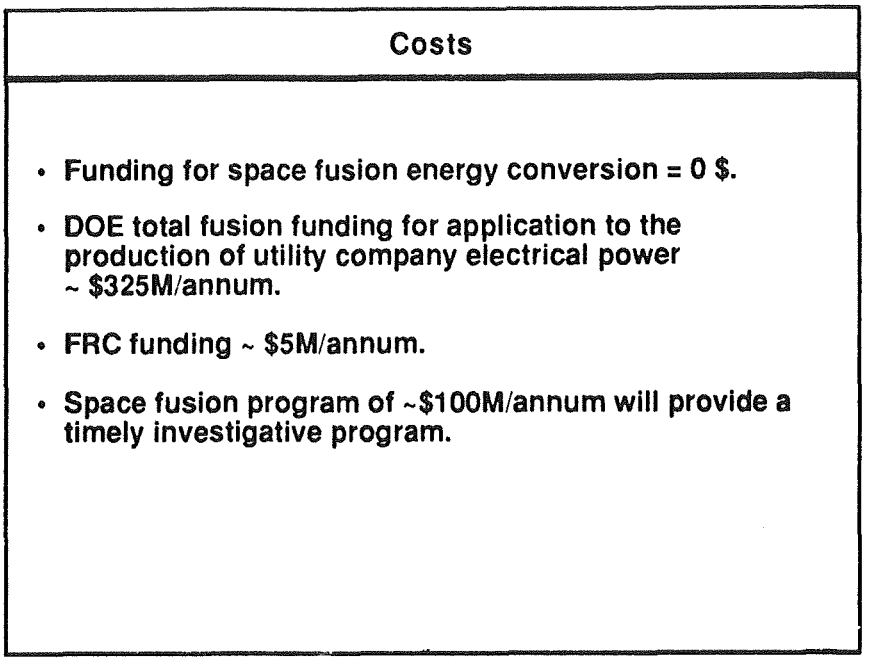


Figure 27. Program cost status/projections for space fusion research.

The best program approach is considered to be to design a series of large step, high risk FRC experiments aimed at quickly demonstrating a space fusion reactor capable of burning D-^{3}He. The plasma is believed to be capable of being heated to ignition using neutral beam injection and of being maintained stable by the beam flux. Experimental verification is required.

This empirical approach, by-passing the depth of understanding desired by a science program, is appropriate for an engineering developmental program and has, in fact, been a path successfully taken to implement prior inventions. This must be accepted as an expedited but high risk approach. The magnitude of the gain to space programs justifies the risk level and warrants the recommendation. It should be emphasized that the cost estimates are no more than educated estimated judgments to demonstrate plasma stability in an FRC. More definitive cost estimating needs to be performed.

SCHEDULE

The anticipated schedule for achieving fusion energy conversion for a FRC program is shown in Figure 28.

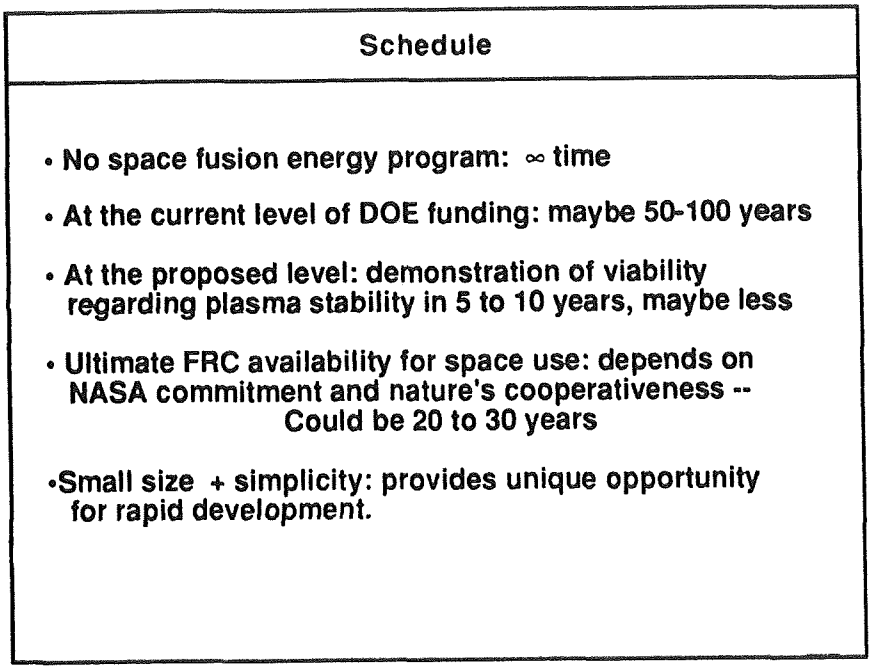


Figure 28. Program schedule status/projections for space fusion research.

KEY POINTS

With reference to the developmental responsibilities of fusion for space, there are several significant points that must be considered, Figure 29. Program success largely depends upon the last point, i.e., NASA has a vested interest.

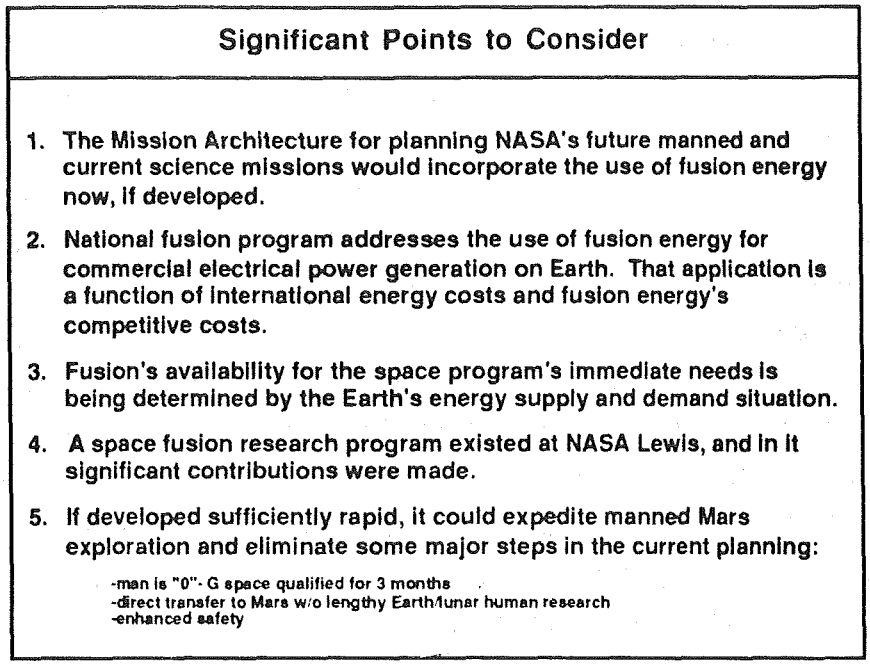


Figure 29. Considerations in undertaking a space fusion program.

Fusion energy can serve as a key element in the mission architecture in accomplishing the "U. S. National Space Policy." That is based upon an excellent matching of fusion's capabilities with the technical requirements that result from the policy -- as discussed in this report's content: "The overall goals of the United States space activities are: ... (2) to obtain scientific, technological and economic benefits for the general population and to improve the quality of life on Earth through space-related activities and to expand human presence and activity beyond Earth orbit into the solar system." ("US National Space Policy," November 2, 1989, p 1 (ANON89)) "The objectives of the United States civil space activities shall be (1) to expand knowledge of the Earth, its environment, the solar system, and the universe; (2) to create new opportunities for use of the space environment through the conduct of appropriate research and experimentation in advanced technology and systems; (3) to develop space technology for civil applications and, wherever appropriate, make such technology available to the commercial sector; (4) to preserve the United States preeminence in critical aspects of space science, applications, technology, and manned space flight; (5) to establish a permanently manned presence in space; and to engage in international cooperative efforts that further United States overall space goals." (ibid. pp 2-3) In order to further and to continue research in space and to conduct manned exploration much beyond Earth orbit will entail the availability of

high energy sources to move large payload masses and to conduct timely missions at greater and greater distances as the lesser energy demanding missions and space goals become fulfilled. The space program will be compelled to incorporate into its space transportation infrastructure more efficient systems that offer quantum leaps in performance rather than minor refinements in the lesser energy intense systems. That will be required for logistical support beyond the Earth-moon space operational regime to achieve the economy necessary for reasonable support of those missions. Fusion energy has the potential for providing that energy source due to its high specific energy release and variable high performance propulsion capability, provided that the technology can be appropriately developed for meeting the space application needs. We recommend leveraging of research funds for high leverage technological payoffs to assure that a US space vision for the future will materialize. Otherwise the space program's energy conversion infrastructure will not be in a position of advancing with the needs of exploration and science research programs.

CONCLUSIONS

Figure 30a and b presents the conclusions of the authors:

Conclusions

1. Fusion energy offers very attractive inherent features for accomplishing space mission requirements and becoming perhaps the key element in the United States Space Mission Architecture for fulfillment of the U. S. National Space Policy.
2. Fusion's application in space is for programs currently being planned and, if available as an element in the space transportation infrastructure, could have been used and incorporated into a more ambitious space science and exploration program.
3. A successful DOE fusion research program will produce fusion reactors useful on Earth, but not for space applications. There is a lack of commitment to space fusion energy conversion.
4. Fusion would greatly enhance safety for manned missions.
5. The space program's launch operational costs for manned logistic flights to Mars - using fusion energy conversion - would be substantially reduced.
6. Fusion energy's high specific power performance advantages will pay for the development costs many times over.

Figure 30a. Conclusions.

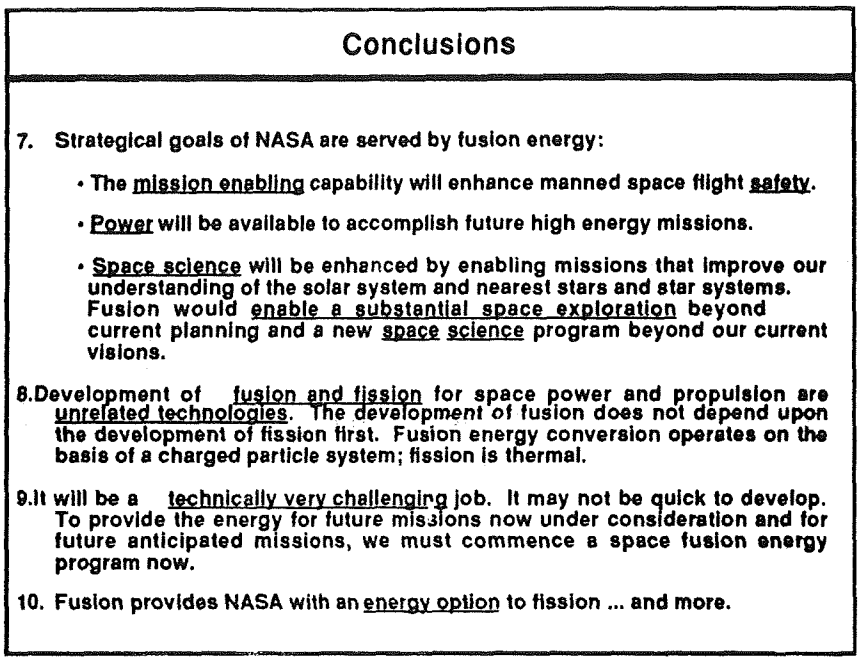


Figure 30b. Conclusions.

RECOMMENDATIONS

Recommendations are provided in Figure 31:

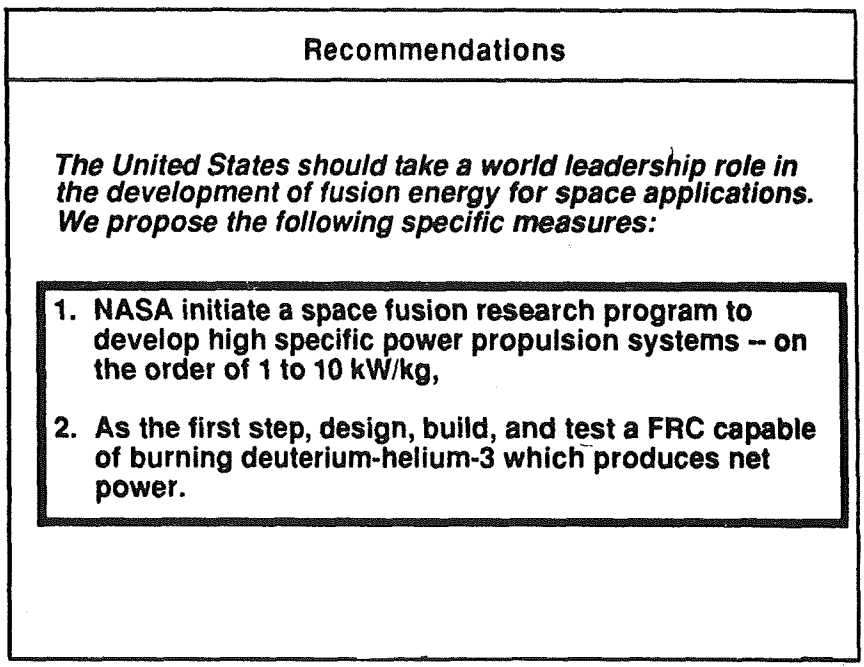


Figure 31. Recommendations.

CONCLUDING REMARKS

The specific fusion reactor concept preferred is the Field Reversed Configuration (FRC). That reactor design approach inherently offers a high beta design; and although it is classified as a compact toroid, its external topology naturally lends itself to the generation of thrust. Burning deuterium and ^{3}He will reduce the neutron flux level substantially and will produce a very large part of the reaction's energy in charged particles for the efficient conversion of plasma energy directly to thrust without the inefficiencies associated with thermal systems. The primary concern with the FRC is plasma stability while operating under net power regimes, and that is a subject which will have to be addressed by full scale experiments. Neutral beam injection into the plasma is proposed to aid in plasma stability and for raising the plasma energy level to ignition. Helium-3 has been determined to be available on the moon in a sufficient quantity to support the space program's fuel requirements for flight programs. Enough ^{3}He is available on Earth now to commence a FRC D-^{3}He reactor experimental test program. One ^{3}He fuel supply option to lunar mining is the proton-lithium-6 reaction at least until the lunar supply becomes available. Fusion energy development is considered to be high risk research, but that risk is considered insignificant in comparison to the enormous benefits that can be realized from energy conversion systems having such desirable properties that enable future space missions.

In summary, a space fusion energy capability is considered to be mandatory for performing space missions which implement the "U. S. National Space Policy." If available, excellent use could be made of fusion energy **now**. With only the present DOE fusion research program -- one intended to produce electrical power for electrical utility companies as a profit making venture, the development of fusion energy for space -- a different application -- will not occur in the foreseeable future unless a major redirection of charter and program focus is mandated. Space fusion energy is considered to be high risk, but extremely high gain, research that must be undertaken by NASA. Otherwise the future of the United States' space program can be expected to stagnate as advanced missions in space become energy constrained in the not too distant future. If the United States does not act, some other country can be anticipated to fill the void by undertaking the development of fusion energy for space.

SYMBOLS

^3He	helium-3, isotope of helium
^{11}B	boron-11, isotope of boron
AU	astronomical unit = $1.5x10^{11}m$
c	velocity of light = $3x10^8m/s$
D	deuterium, isotope of hydrogen
E	energy
GW	gigawatts (10^9 watts)
I_{sp}	specific impulse, seconds
J	energy, joules
keV	kiloelectron volts
kg	kilograms
m	mass
m	meters
MeV	million electron volts
M_o	initial vehicle mass, MT (= propellants + inert vehicle + payload)
M_p	propellant mass, MT (includes fuels and diluent)
MT	metric tons
MW	megawatts
n	ion density, number of ions per cubic centimeter
n	neutron
$n\tau$	Lawson parameter, $cm^{-3}s$ (fusion plasma = plasma losses)
N	thrust, newtons
p	proton
P_j	jet power, kW
s	seconds
s	gyroradius, cm, (characteristic radius of a charged particle's orbit gyrating around field lines in a magnetic field)
t	flight time

T	temperature.°K
T	tritium, isotope of hydrogen
T_i	plasma's ion temperature, °K or keV

Greek

α_p	propellant system specific power, kW/kg
α_{p1}	propellant system specific power where α_p=1kW/kg
α_{p10}	propellant system specific power where α_p=10kW/kg
β	ratio of plasma pressure to magnetic field pressure, %
Δv	incremental velocity change, km/s
γ	payload mass fraction, % (payload mass/initial vehicle mass)
σ	nuclear cross section, cm^2
$<\sigma v>$	reactivity parameter, cm^3/s
τ	fusion reaction time, seconds

ACRONYMS

FRC	Field Reversed Configuration, magnetic confinement experiment
HESM	High Energy Space Mission
JET	Joint European Torus, magnetic confinement experiment
LEO	Low Earth Orbit
NERVA	Nuclear Engine for Rocket Vehicle Application (fission thermal rocket)
PLT	Princeton Large Torus, magnetic confinement experiment
TFTR	Tokamak Fusion Test Reactor,

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PRESENTATION 1.4.4

ADVANCED PROPULSION CONCEPTS

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ADVANCED PROPULSION CONCEPTS

Presentation to the
Space Transportation Propulsion Technology Symposium
Pennsylvania State University
June 27, 1990

JPL

Robert H. Frisbee, Ph.D.

Propulsion Systems Section
Jet Propulsion Laboratory
California Institute of Technology

ADVANCED PROPULSION CONCEPTS

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Robert H. Frisbee, Ph.D.
Jet Propulsion Laboratory
California Institute of Technology
Pasadena California

This work described in this presentation was performed by the Jet Propulsion Laboratory, California Institute of Technology, under contract to the National Aeronautics and Space Administration.

OUTLINE

This presentation will discuss a variety of Advanced Propulsion Concepts (APC). The focus will be on those concepts that are sufficiently near-term that they could be developed for the Space Exploration Initiative (SEI) on a time scale consistent with the President's call for a return to the Moon and a landing on Mars by July 20, 2019.

Several other advanced concepts, such as nuclear thermal propulsion and megawatt-class electric propulsion, have been presented earlier; this presentation will discuss high-power (multi-megawatt) electric propulsion, solar sails, tethers, and extraterrestrial resource utilization concepts. This will be followed by a summary of these concepts and some general conclusions on their technology development needs.

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OUTLINE

- Introduction
- Focus on Near-Term Advanced Propulsion Concepts That Can Be Developed for the Space Exploration Initiative :
 - Multi-Megawatt Electric Propulsion
 - Solar Sails
 - Tethers
 - Extraterrestrial Resource Utilization
- Summary and Conclusions

INTRODUCTION

As a general definition, Advanced Propulsion Concepts (APC) are those propulsion concepts beyond advanced chemical (e.g., O_2/H_2) propulsion. These advanced concepts hold the promise of significantly benefiting the Space Exploration Initiative (SEI) missions of the 21st Century. However, other than the very near-term nuclear thermal propulsion and megawatt-class electric propulsion concepts discussed previously, these APCs discussed here will require significant further research in order to resolve issues relating to feasibility, performance, or mission benefit. Depending on the maturity of a given concept, the required research can range from proof-of-principle experiments for far-term concepts to experiments designed to quantify performance parameters (e.g., specific impulse, efficiency, thruster lifetime, etc.) for the more near-term concepts. Finally, note that although most of the mission applications discussed in this presentation will be for the piloted lunar and Mars missions, these APCs can also be used for a number of the unmanned precursor missions of the SEI. More generally, APCs can be applied to a variety of ambitious unmanned missions such as outer-planet orbiters, sample returns, or interstellar precursor missions.

When assessing the missions benefits of APCs, the two primary figures of merit that are typically used are the total transportation system initial mass in low Earth orbit (IMLEO) and the mission trip time. The total IMLEO can include the "dry" mass of the vehicle (engines, propellant tanks, etc.), its propellant, a propellant "tanker" if the propellant is launched separately from the vehicle, on-orbit constructed or support facilities (e.g., space station), and, finally, the payload. Often IMLEO is used as the primary figure of merit since it directly relates to the launch costs for transporting materials from the Earth to low Earth orbit (LEO). In general, savings in IMLEO, and thus launch costs, for an advanced concept (as compared to a state-of-the-art system) are used to offset the development costs of the advanced system.

One interesting result of mission trade studies of APCs is that their benefit is a function of the mission size; in general, the "bigger" the mission, the more the benefit of the APC. This is one reason why APCs are often considered for the large piloted missions of the SEI. This behavior is seen because, in general, state-of-the-art (SOA) systems have a small fixed mass (e.g., dry mass) as compared to the APC system; however, the propellant mass required for the SOA system increases with increasing mission "size" (payload mass, Delta-V) more rapidly than for the APC system due to the higher specific impulse (I_{sp}) of the APC system.

Trip time can also be an important factor in assessing the mission benefits of APCs, since the longer the trip time, the higher the operations costs, and the higher the required system reliability and lifetime. Also, for missions like a lunar base buildup, trip time can impact the vehicle fleet size if the trip time is too long to allow re-use of the vehicle on the required delivery schedule. Finally, trip time is especially important for piloted missions because of the effects of long exposure of humans to weightlessness or radiation, or to the psychological effects of long-duration missions. This is especially important in piloted Mars missions, since most high-thrust (ballistic) missions have trip times of several years. As will be discussed below, some low-thrust APCs can reduce this trip time to one year or less.

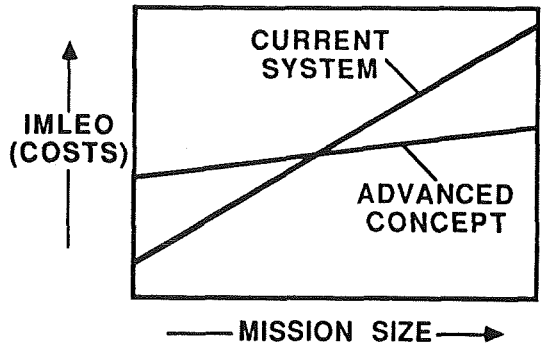
In general, for missions in cis-lunar space, high-thrust propulsion system, such as chemical or nuclear thermal, generally have shorter trip times than low-thrust systems, such as electric propulsion. However, for missions beyond the Moon, low-thrust APCs, which can thrust continuously for days to years, can have a shorter trip time than high-thrust systems which must coast ballistically to their target. Thus, APCs often show a trip time benefit only for missions beyond the Moon.

Finally, other factors that can be of interest include the schedule requirements (i.e., can the APC be developed in time to meet the mission schedule), nuclear safety (and its impact on operations and nearby vehicles), and development costs. In the case of development costs, synergisms may exist between different agencies; for example, the technology of the nuclear power system required for a high-power nuclear electric propulsion vehicle is of interest to the National Aeronautics and Space Administration (NASA), Department of Energy (DoE), and Department of Defense (DoD).

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INTRODUCTION

- Advanced Propulsion Concepts (APC) Are Those Concepts Beyond Advanced Chemical
 - Promise Significant Mission Benefits for the Space Exploration Initiative (SEI) Missions of the 21st Century
 - Require Further Research to Demonstrate/Quantify Performance
- Primary Figures of Merit Used in Assessing APCs Are the Total Transportation System Initial Mass in LEO (IMLEO) and Trip Time
 - IMLEO -> Launch Costs
 - Trip Time -> Ops. Costs, Reliability, Fleet Size, Life Sciences
 - Other Factors Include Risk (Schedule), Nuclear Safety, and Development Costs



ADVANCED PROPULSION CONCEPTS

As discussed previously, the focus of this presentation will be on those Advanced Propulsion Concepts (APC) that can be developed in time to support the piloted lunar and Mars SEI. The most near-term APCs, discussed in previous presentations, are nuclear thermal propulsion and electric propulsion, including both solar electric propulsion (SEP) and nuclear electric propulsion (NEP), at power levels up to a few megawatts. High-power (>10 MW_e) SEP and NEP, solar sails, tethers, and extraterrestrial resource utilization for propellant production will be discussed in detail in this presentation.

There are a wide variety of other Advanced Propulsion Concepts that are not being discussed because they are more far-term, require a large on-orbit infrastructure, or do not provide major benefits to the SEI in terms of IMLEO or trip time. For example, exotic chemical propellants, such as atomic or free-radical hydrogen, may greatly enhance launch vehicle performance, but they should be considered far-term since significant research is required at this point to even demonstrate feasibility. Similarly, fusion and antimatter propulsion, which may enable very fast trips in the solar system (e.g., a two-month round trip to Mars with fusion propulsion), are also far-term. Laser or microwave beamed-energy concepts are applicable to only cis-lunar space (because of optics transmission range limitations) and may require a large on-orbit infrastructure (e.g., laser power stations, relay mirrors, etc.). Solar thermal propulsion, rail guns, and mass drivers may provide significant reductions in IMLEO for cis-lunar operations but only modest reductions in IMLEO for Mars missions. Also, since they are low-thrust systems, they will have a longer trip time for cis-lunar missions than high-thrust concepts.

Solar thermal propulsion is a very near-term propulsion concept (under development by the Air Force) that is especially suited to cis-lunar missions; it has many of the specific impulse (I_{sp}) advantages of nuclear thermal or laser thermal propulsion, but without the nuclear reactor or laser system infrastructure of the latter two. Similarly, rail guns and mass drivers have several benefits for operations in cis-lunar missions. Although they are essentially electric propulsion concepts, they can use any material as "propellant"; thus, lunar-produced materials (including raw lunar soil) could be used as propellant, thereby greatly reducing the required IMLEO. However, rail guns and mass drivers should be considered as far-term concepts, in part because of the need for a pre-established lunar materials production infrastructure. There is also the need for significant technology development of the thrusters, although some of this technology is being addressed by DoD programs.

JPL ADVANCED PROPULSION CONCEPTS

- **Focus on Near-Term Advanced Propulsion Concepts (APC) Likely to Have an Impact on the Lunar and Mars SEI**
 - **Nuclear Thermal***
 - **Solar and Nuclear Electric Propulsion (MW* & MMW)**
 - **Solar Sails**
 - **Tethers**
 - **Extraterrestrial Resource Utilization**
- **Other APCs Probably More Far-Term, Require Significant On-Orbit Infrastructure, or Don't Satisify Both Lunar and Mars SEI**
 - **Exotic Chemical (Atomic Hydrogen)**
 - **Beamed Energy (Laser, Microwave)**
 - **Solar Thermal**
 - **Rail Gun / Mass Driver**
 - **Fusion**
 - **Antimatter**

* Discussed in Previous Papers

MULTIMEGAWATT ELECTRIC PROPULSION

Multimegawatt (>10 MW_e) SEP and NEP have the potential for both reducing IMLEO and trip time. For example, in a split Mars mission, where the cargo is sent on a slow, low-energy trajectory and the piloted vehicle is sent on a fast, high-energy trajectory, NEP and SEP cargo vehicles operating at tens of megawatts of power have an IMLEO one-third that of a chemical (O_2H_2) aerobraked vehicle. This large savings in IMLEO (about 1000 metric tons for a 400 metric ton payload) is offset somewhat by the long trip time of the low-thrust SEP and NEP cargo vehicles as compared to the high-thrust chemical vehicle (600 to 700 days at 10 MW_e versus ~290 days Earth-to-Mars trip time, respectively), although the electric propulsion vehicles are returned to Earth for later re-use.

However, the primary advantage of high-power electric propulsion is that it can provide the short round-trip times that may be mission enabling for the piloted portion of a Mars mission. For example, at power levels of 100 to 150 MW_e, an NEP vehicle can achieve a one-year round trip to Mars. The potential for high-power NEP to enable this short trip time is very important, since the round trip time of high-thrust ballistic trajectories (typically two to three years) far exceeds U.S. or Soviet continuous manned experience in space. The long trip times required for piloted Mars missions raise serious health and safety issues. Most notable among these is the problem of long periods of weightlessness (bone and muscle mass loss, etc.). Even if artificial gravity is employed, there still remain the risks associated with prolonged radiation exposure (cosmic rays or solar flares) and the psychological impacts of confinement in small isolated groups. These problems can be accommodated for trip times of one year or less based on the success of Soviet long-duration space station missions.

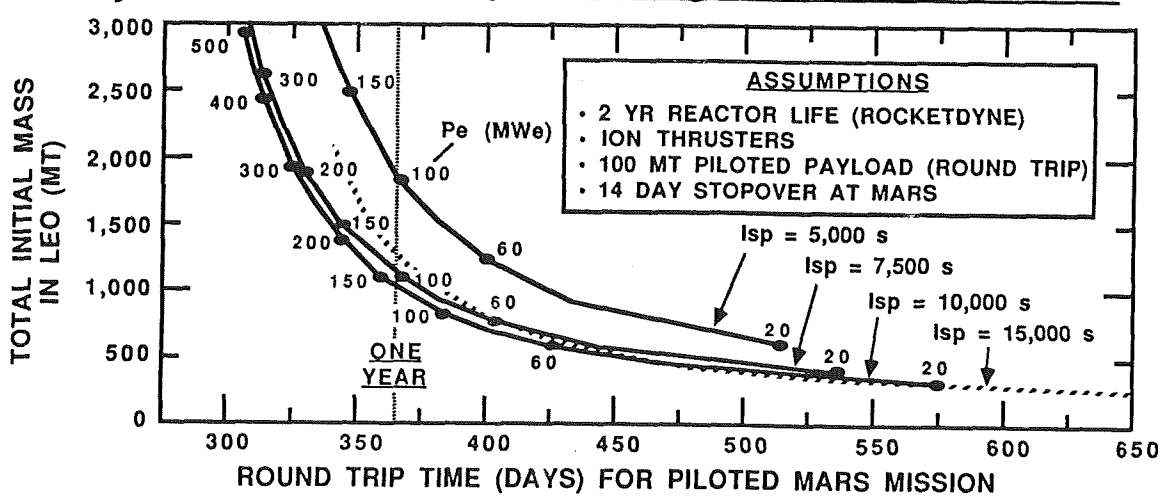
Finally, both SEP and NEP are candidates for the cargo mission. The specific mass of the NEP system is generally less than that of the SEP system in the multimegawatt range, so the NEP vehicle is lighter. However, the NEP vehicle requires an orbit transfer vehicle (OTV) to operate from LEO to a nuclear safe orbit (NSO), typically 1000 km in altitude, where the NEP vehicle operates. This infrastructure overhead, although small, results in the NEP system having roughly the same IMLEO as the SEP system for a given power level, although the NEP vehicle will be somewhat faster than the SEP vehicle due to its lower mass and constant power output.

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MULTIMEGAWATT ELECTRIC PROPULSION

- **Potential for Both Reduced IMLEO and Short Trip Times for High Power (> 10 MWe) Electric Propulsion**

⇨ **Short Trip Times May Be Enabling for Piloted Mars Missions**



- **MMW Solar Electric Propulsion (SEP) and Nuclear Electric Propulsion (NEP) Are Both Options**
 - **MMW NEP Has Lower Specific Mass Than SEP, but NEP Requires Nuclear Safe Orbit (NSO) Operation and LEO-to-NSO Transport Infrastructure**

MMW NEP NUCLEAR ELECTRIC POWER SYSTEMS

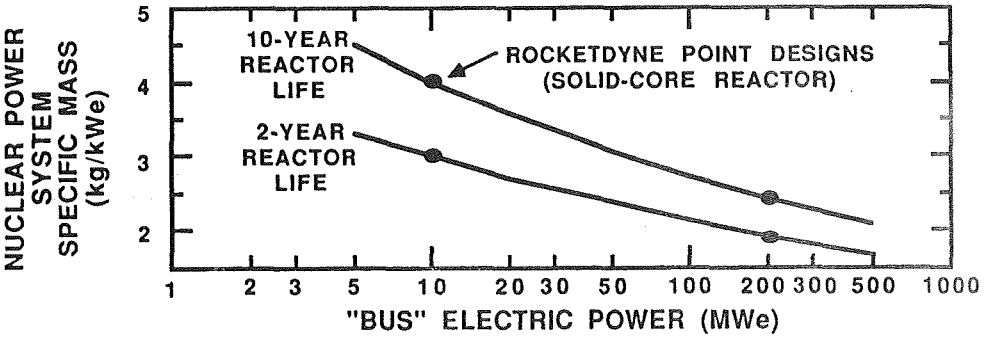
The performance of multimegawatt (MMW) NEP vehicles depends critically on the specific mass of the nuclear electric power system. For the MMW power regime (>10 MW_e), high-temperature dynamic power conversion systems provide significant mass savings over the static power conversion systems favored at low power level (<1 MW_e). There can also be significant economics of scale at the higher power levels since many of the fixed-weight components become a small fraction of the total system mass. For example, the figure illustrates the power system specific mass of a solid-core reaction system with a dynamic Rankine power conversion cycle. These values are taken from an on-going study by Rocketdyne addressing system concepts for MMW steady state Strategic Defense Initiative (SDI) applications. This is one of several concepts being investigated by the DoE for the Strategic Defense Initiative Office (SDIO); there is a strong synergism between these applications and the NEP systems discussed here.

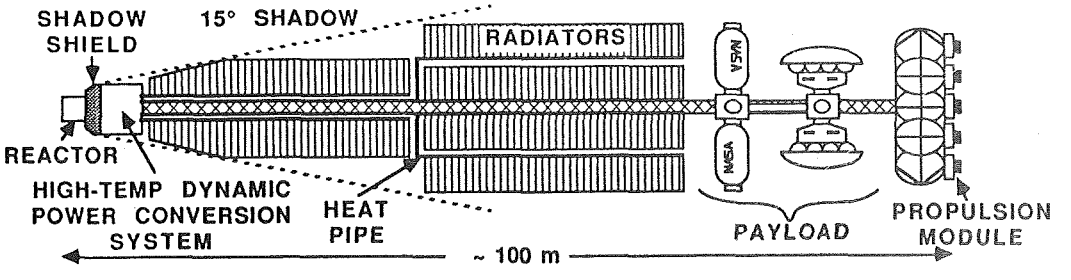
This figure illustrates the effect of power level on specific mass; for example, the specific mass varies from 3.0 kg/kW_e at 10 MW_e to 1.9 kg/kW_e at 200 MW_e. As with many space power concepts, the system mass and the vehicle configuration are dominated by the waste-heat radiators. The vehicle is configured such that the radiators, payload, and propulsion system are behind the reactor's shadow shield. The payload is placed far away from the reactor to minimize radiation dosage, and the thrusters are oriented so as to not impinge the exhaust plume on the radiators and possibly heat them or deposit materials that could change the emissivity of the radiators.

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MMW NEP NUCLEAR ELECTRIC POWER SYSTEMS

- High-Power (> 10 MWe) Space Nuclear Power System
 - High-Temperature Dynamic Power Conversion System
 - Significant Economies of Scale
- Example : Solid-Core Reactor & Dynamic Rankine Cycle Based on Rocketdyne MMW Steady-State SDI Systems Concepts



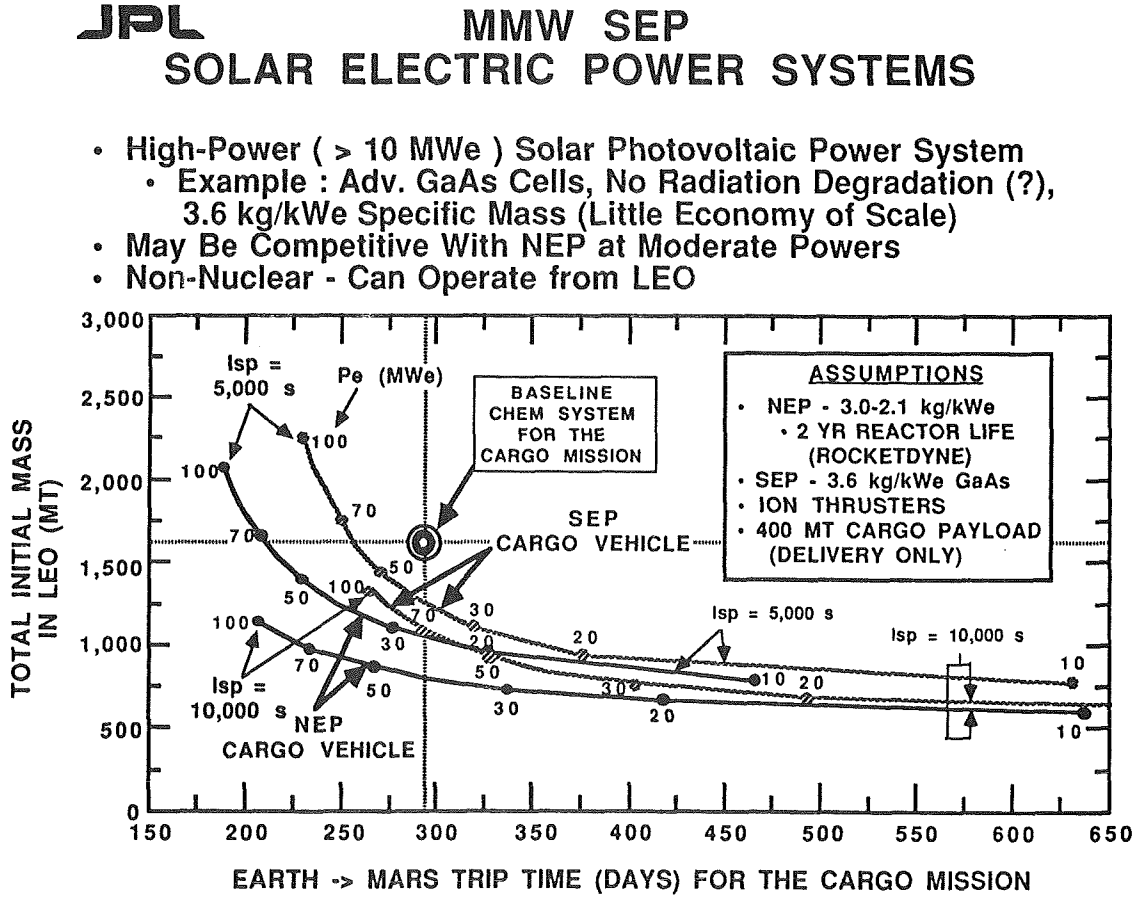


MME SEP SOLAR ELECTRIC POWER SYSTEMS

As mentioned earlier, MMW SEP systems can compete with MMW NEP for the Mars Cargo mission, as shown in this figure. Since it is non-nuclear, an SEP vehicle can operate directly from LEO, thereby avoiding the infrastructure overhead associated with an NEP vehicle. Also, advanced solar photovoltaic power systems may be directly competitive with nuclear power systems. For example, for the SEP system shown here, the specific mass of the power system is 3.6 kg/kW$_e$; based on the Rocketdyne study discussed above, a nuclear power system with a 10-year operating life would have the same specific mass at a power of 20 MW$_e$. However, the nuclear system does provide constant power whereas the solar power system's output drops off as it moves away from the sun. Interestingly, because efficiency of the GaAs photovoltaic cells assumed in this example increases with decreasing temperature, the power output is slightly better than a simple $1/R^2$ distance relationship. Finally, note that it is both the lower specific mass and the constant power output that gives high-powered NEP the advantage in trip time over SEP.

One potential drawback of MMW SEP is the lack of significant economies of scale at higher powers due to the modular nature of the solar photovoltaic arrays. In fact, there may even be negative economies of scale due to added structural complexity (e.g., active structure control) or increased mass and losses in the transmission lines. The latter effect is due to the relatively low voltage output of the solar arrays (several hundreds of Volts), resulting in the need for large bus bars at high powers. By contrast, the power output from a nuclear power system with dynamic power conversion can be at high voltages (several thousands of Volts), thereby minimizing transmission line losses and mass. This issue needs to be addressed in future studies.

Another issue relating to solar photovoltaic power systems is the impact of radiation degradation on the cells due to passage through the Van Allen radiation belts. Other studies have shown that a single round trip can result in as much as a 50% degradation in power output; however, in the example shown here, no radiation degradation was assumed based on the assumptions of a fast trip time through the radiation belts coupled with techniques for minimizing cell damage (e.g., high-temperature annealing, radiation-resistant materials, etc.). The issue of radiation degradation, and its impact on vehicle performance, will continue to be an area requiring further technology development and mission analysis.



MMW EP THRUSTERS

Various types of high-power electric propulsion thrusters are in development. For MMW SEP or NEP applications, it is very desirable to have thrusters that can operate at high power levels (1 MW_e or more) per thruster, so as to minimize the number of thrusters required and thereby reduce system complexity. High specific impulses (5,000 to 10,000 lb_f-s/lb_m) are needed. The optimum I_{sp} depends on the mission (Delta-V) and vehicle (specific mass, thruster efficiency, etc.), although values of I_{sp} in excess of 10,000 lb_f-s/lb_m tend to increase the mission trip time while only slightly reducing the IMLEO. It is also important that the thrusters have a high electric-to-jet power efficiency so as to maximize thrust per unit power as well as reduce thermal control requirements. Finally, a long thruster lifetime is desirable to minimize the number of spare thrusters needed to complete the mission. Unfortunately, there is no single type of thruster that meets all of these requirements; each of the thfusters discussed below has different advantages and disadvantages.

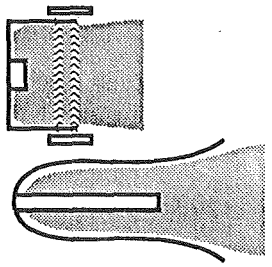
The two most near-term electric thrusters are the ion thruster and the self-field magnetoplasmadynamic (MPD) thruster. Ion thrusters currently operate at levels of one to ten kilowatts per thruster; advanced ion engines may be capable of a megawatt per thruster. By contrast, MPD thrusters begin to operate efficiently at powers of about a megawatt (or more) per thruster. Both ion and MPD thrusters can operate at high I_{sp}, although low molecular weight propellants (e.g., H_2) may be required for MPD thrusters operating at an I_{sp} of 10,000 lb_f-s/lb_m. The efficiency and lifetime of ion thrusters (70% to 90% and tens of thousands of hours, respectively) are greater than that of MPD thrusters (40% to 60% and hundreds to thousands of hours, respectively). Thus, there is no clear winner between ion and MPD thrusters; development of both types must continue since one may be favored over the other for some classes of missions, but not for all.

Several other advanced electric thrusters are currently in the research stage. Two examples of these are the electron cyclotron resonance (ECR) thruster and the ion cyclotron resonance (ICR) thruster. In both of these, microwave energy is used to excite and energize the propellant to produce a high-energy plasma which is controlled and directed by externally applied magnetic fields. The more near-term of the two, the ECR thruster, couples the microwave energy to electrons in the plasma. The ECR thruster is potentially scalable over a range of kilowatts to many megawatts per thruster. It can operate at moderate-to-high efficiency (50% to 80%) and high I_{sp}. Interestingly, because it is an electrodeless device (unlike ion or MPD thrusters), it has the potential for very long thruster lifetimes. Also, because it is an electrodeless device, the ECR thruster can use a variety of propellants, including oxidizing propellants such as oxygen derived from extraterrestrial resources.

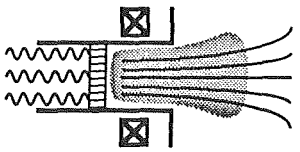
In the ICR thruster, under development at MIT, microwave energy is coupled to ions in the plasma. This far-term thruster concept would operate at many megawatts per thruster at high I_{sp} (with hydrogen propellants), and at a moderate efficiency (50%). One unique feature of the ICR thruster concept is its ability to continuously vary I_{sp} over the course of a mission so as to optimize vehicle performance (i.e., lower I_{sp} and thus higher thrust for planetary escape or capture followed by higher I_{sp} for the heliocentric transfer). Although most electric propulsion thrusters can also vary their I_{sp}, they typically do so with some loss in performance (e.g., efficiency). The ICR thruster is designed to permit easy and efficient variation in I_{sp} to meet the needs of the mission trajectory.

JPL MMW EP THRUSTERS

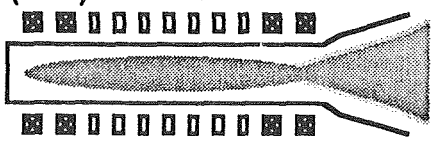
- Various Types of High Power Electric Thrusters in Development
 - Need MW per Thruster
 - High Isp (5,000 - 10,000 lbf-s/lbm) Desirable (Mission Dependant)
 - High Eff. and Long Life Desirable
 - Ability to Use Common or ET-Resource Derived Propellants Desirable
- Near-Term : Ion and MPD Thrusters
 - Ion : Low-to-Medium Power/Thruster, High Isp, High Eff., Medium-to-Long Life
 - MPD : High Power/Thruster, High Isp (w/ H2), Medium Eff., Medium Life (Electrode Erosion)



- Mid-Term : Electron Cyclotron Resonance (ECR) Thruster
 - Microwave Energy Coupled to Electrons in Plasma
 - Low-to-High Power/Thruster, High Isp, Medium Eff., Can Use O2, etc.
 - Long Life (Electrodeless)



- Far-Term : MIT Ion Cyclotron Resonance (ICR) Thruster
 - Microwave Energy Coupled to Ions in Plasma
 - High Power/Thruster, High Isp, Medium Eff., H2 Propellant



MMW EP SUMMARY

Multimegawatt electric propulsion, unlike most near-term Advanced Propulsion Concepts, has the potential for reducing both the IMLEO and trip time for ambitious missions of the SEI. In particular, MMW NEP, because of its ability to provide short trip times, may be enabling for the piloted Mars mission. In terms of overall mission performance, an MMW NEP system is lighter and faster than a similar-power SEP system, but the NEP system does have the added overhead and complexity of nuclear operations and the corresponding infrastructure required to support space-based nuclear power systems. Finally, although 100 MW_e class NEP vehicles are required for piloted Mars missions, NEP or SEP vehicles at power levels of a few tens of megawatts are attractive for the cargo mission where reductions in IMLEO, rather than trip time, are more important.

Current technology programs are addressing several of the technologies of interest here. However, in several cases, only low power applications are being pursued; these programs will need to be extended to cover the MMW regime. The DoE is evaluating several MMW nuclear power system concepts for the SDIO. Steady-state systems (rather than burst-mode systems) are of special interest to the MMW NEP vehicle concept. For SEP, both the Lewis Research Center (LeRC) and the Jet Propulsion Laboratory (JPL) are investigating high-power solar arrays. Both ion and MPD thrusters are being developed at LeRC and JPL, although little work has been done on megawatt-class ion thrusters. Finally, the two advanced thrusters discussed earlier are in the basic research stage; the ECR thruster is being studied at JPL and Caltech, the ICR thruster at MIT.

Technology needs for MMW EP include those of large space structures, power, and thrusters. For the nuclear systems, the required technologies include MMW reactors, dynamic power conversion systems, and lightweight radiators. Note that there may be some significant differences between power systems designed for SDI applications and NEP vehicle applications, such as in the area of "hardness" or vulnerability. In the area of solar power systems, there is a need for lightweight, high-efficiency, radiation-degradation resistant cells and substrates. High-temperature, high-efficiency cells are also important in the laser-electric propulsion concept, in which a laser beam (rather than sunlight) is used to power an SEP-type vehicle.

For ion thrusters, there is a need to develop thrusters with a high power (megawatts) per thruster. There is also a need to demonstrate high-I_{sp} and high-efficiency operation of ion thrusters using common propellants such as argon or krypton, since the xenon currently used is very expensive and may not be available in the quantities required for the Mars SEI. There is also a need to demonstrate high power-per-thruster operation of MPDs, although this is currently more a facilities limitation than a thruster limitation since the MPD is intrinsically a high-power device. Also, MPDs are currently limited in their lifetime due to erosion of the electrodes; both lifetime and efficiency need to be improved. The high I_{sp} (up to 10,000 lb_f-s/lb_m) that may be required for Mars missions will require development of MPDs that can operate on low molecular weight propellants such as hydrogen. Finally, advanced thruster concepts, such as the ECR and ICR thrusters, require continuing basic research to demonstrate and characterize their performance.

JPL MMW EP SUMMARY

- Potential for Both Reduced IMLEO and Short Trip Time
 - Short Trip Times May Be Enabling for Piloted Mars Mission
- Mission Benefits Issues :
 - High Power (100 MWe Class) NEP Needed for Piloted Mars Missions
 - High Power NEP Lighter & Faster Than Similar Size SEP, but NEP Requires Nuclear Operations & Infrastructure
 - Medium Power (10-100 MWe) Attractive for Cargo Missions
 - NEP and SEP Both Contenders
- Current Work :
 - Nuclear : DoE/SDIO
 - Ion : JPL and LeRC
 - ECR : JPL/Caltech
 - Solar : LeRC and JPL
 - MPD : JPL and LeRC
 - ICR : MIT
- Technology Needs :
 - Large Space Structures
 - Nuclear : MMW Reactor, Dynamic Conversion, Radiators
 - Solar : Lightweight, High Eff., Radiation Resistant Cell Blankets
 - Ion : High Power/Thruster, Ordinary Propellants
 - MPD : Lifetime (Erosion), Eff., High Isp
 - ECR, ICR : Basic Research to Demonstrate/Characterize

SOLAR SAILS

Solar sails operate by using momentum exchange with solar photons; this amounts to a force of 9 Newtons/km^2 at 1 AU. As such, a solar sail has "infinite" specific impulse, because it requires no propellant, but it has a low acceleration resulting in long trip times. Also, solar sails are typically large, gossamer structures with dimensions of kilometers; for example, a typical solar sail has an area of 4 km^2. Because of their light weight and "infinite" I_{sp}, solar sails represent the lightest advanced propulsion concept. Solar sails are also potentially one of the most near-term of the APCs, having been extensively analyzed in the past. The primary disadvantage of solar sails is their low acceleration, which results in very long trip times. Thus, solar sails are suited only to cargo missions, although for these missions they can provide major reductions in IMLEO.

From a mission performance perspective, the long trip times of solar sails represent a significant issue which must be resolved by mission planners. For example, the long trip times affect not only the sail's lifetime requirements, but also the storage life requirements of the cargo. Scheduling of departure/arrival dates may also be complicated by the long trip times.

One way to reduce the trip time is by eliminating the long planetary escape/capture spirals by basing the sail at a high altitude, but this then requires an infrastructure of OTVs to ferry cargo from LEO (or low Mars orbit) to the sail's orbit. In fact, this is required at Earth since a sail cannot achieve sufficient thrust to overcome air drag at altitudes less than about 2000 km. Thus, the IMLEO shown in the figure includes the OTV infrastructure required to transport sails and cargo from LEO to the sail's operational altitude (2000 km).

Another significant issue affecting the sail's performance is its method of construction, since this affects the average areal density (grams per square meter) and ultimate acceleration of the sail. For example, current sail technology, studied extensively by JPL for a Halley's Comet rendezvous mission, involves the use of sails which would be deployed (un-folded) in orbit. This requires the use of a relatively thick sail and heavy support structure (booms, etc.), with a fairly high areal density (5 g/m^2), to survive folding on the ground, packaging for launch, and un-folding in orbit. If, however, the sail is assembled in orbit, the sail film and support structure need not be as thick or heavy, since they do not need to survive the folding and un-folding of a sail assembled on the ground. This can result in a roughly five-fold reduction in areal density, but only with the addition of the additional infrastructure of a sail construction facility in LEO.

Finally, if the sail material is fabricated in orbit, zero-gee manufacturing techniques can be used to produce ultra-thin materials and provide a twenty-five fold reduction in areal density as compared to a deployable Halley's Comet class sail. However, although deployable and assemblable Halley's Comet class sails are near-term, sails fabricated in orbit are mid- to far-term, since the technology for zero-gee sail fabrication is yet to be developed.

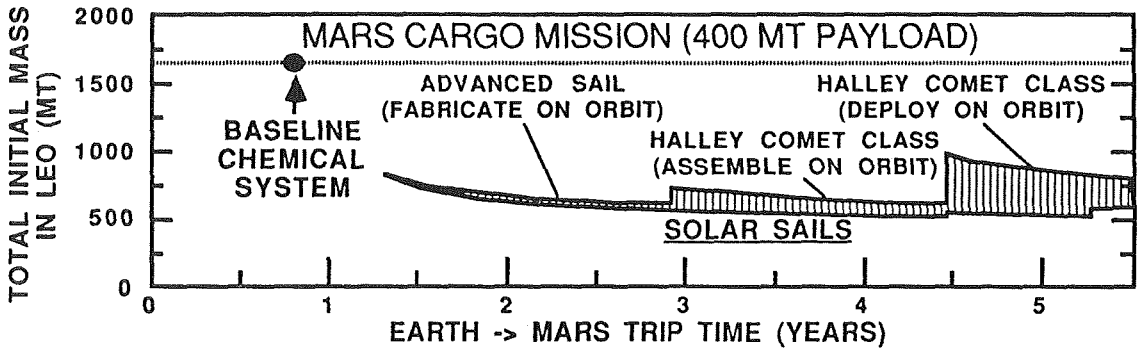
As mentioned earlier, deployable sails were studied extensively by JPL in the late 1970's for use in a Halley's Comet rendezvous mission. The only current solar sail technology work being pursued is by the World Space Foundation (WSF), a private organization. This group has built upon the work done by JPL, using many of the engineers involved in the earlier JPL activity. The WSF has constructed an engineering prototype square sail of 880 m^2; they are in the process of securing space on a launch vehicle to perform a demonstration flight of their prototype sail.

In the area of technology needs, deployable solar sails are relatively mature. There are still issues of dynamics and control of large space structures to be resolved. These issues would need to be resolved for two types of potential sails: the square and the heliogyro sail, both of which were studied by JPL. The square sail is simply a square film of sail material supported by booms and guy wires; the heliogyro sail operates like a propeller with "propeller blades" of sail material unrolled from a central hub and stabilized by the centrifugal force of the spinning system. Both types of sails have different advantages and disadvantages that must be resolved by further study.

A second area of technology need is the development of on-orbit assembly techniques, since assemblable sails provide such significant performance advantages over deployable sails. Finally, in the far-term, the technology of on-orbit zero-gee fabrication of sails from advanced materials will be required to realize the ultimate in solar sail benefits.

SOLAR SAILS

- Reduces IMLEO Since Infinite Isp, but at Cost of Long Trip Times
 - Lightest Near-Term APC (Interplanetary Supertanker)



- Mission Benefits Issues :
 - Long Trip Time (Lifetime)
 - Large Structure Deployed Versus Assembled On-Orbit
 - Impacts Sail Areal Density -> Trip Time

- Current Work :
 - Extensively Studed by JPL for Halley Comet Mission (1978-79)
 - World Space Foundation Engineering Prototype Sail (880 m2)

- Technology Needs :
 - Dynamics / Control (Square Versus Heliogyro Sails)
 - On-Orbit Deployment Versus Assembly Versus Fabrication
 - Materials for Advanced Sails

TETHERS

Tether concepts for propulsion and power have been investigated within the last decade for a variety of space missions. Two classes of tether systems are electrodynamic tethers, which interact with a planetary magnetic field, and non-conducting tethers which interact with the gravitational field. The latter type, which can be used for orbit raising and lowering or planetary escape and capture will be discussed below.

Tethers can reduce IMLEO for the lunar and Mars SEI by reducing or eliminating the propellant required for propulsive maneuvers. For example, rotating tethers can be used in a transportation system in cis-lunar space that requires no propellant. In this system, the orbital angular momentum lost by transferring cargo "up" out of the Earth's gravity well from LEO to the Moon is balanced by shipping lunar materials "down" to LEO. (The rotating tether in lunar orbit rotates at a speed such that its tip has zero velocity relative to the lunar surface, so that payloads can be dropped off or picked up on each cycle.) For Mars missions, tethers can be employed on Deimos and Phobos to lower incoming traffic to lower orbits, or to raise outgoing traffic to higher orbits. In fact, a 6100-km long tether on Deimos can provide an outbound vehicle with Mars escape velocity.

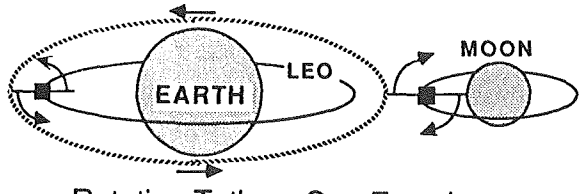
In terms of mission performance, one of the key issues associated with tether systems is the mass of the tether "stations" used to raise or lower the tether for orbital deployment/retrieval. These "stations" typically consist of a power system, tether reel drive motors, structure, and so on. Also, if the station is in a free orbit, rather than based on a moon, propellant is required to re-position the tether station in the proper orbit after each cycle. For example, the tether station would drop to a lower orbit after deploying a spacecraft outward. Typically, the infrastructure represented by the tether stations, their re-boost propellants, as well as the mass of the vehicles used to transport the stations to their final basing location (i.e., lunar orbit, Martian moons, etc.) must be amortized over many payload delivery cycles in order to show a benefit in IMLEO.

Tethers can be used for a variety of applications. One system currently in development for a Space Shuttle flight in the late 1990's will use a downward-deployed tether to lower a probe into the upper atmosphere. For this mission, as well as most of the applications currently considered, no advanced materials technology is required. State-of-the-art materials like Kevlar or Spectra have the physical properties required for tethers in cis-lunar or Martian space. One area that does require further work is that of the dynamics and control of large, flexible systems like tethers.

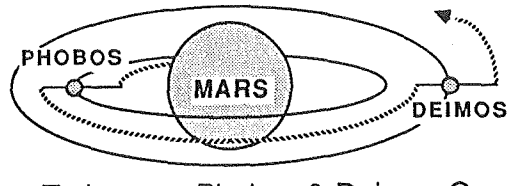
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TETHERS

- **Reduces IMLEO by Reducing / Eliminating Propulsive Maneuvers**
 - **Orbit Raising / Lowering and Planetary Escape / Capture**



- Rotating Tethers Can Function as Cis-Lunar Transportation System That Needs No Propulsion



- Tethers on Phobos & Deimos Can Capture / De-Orbit Landers and Pick-Up / Inject Return Vehicles

- **Mission Benefits Issues :**
 - **Infrastructure Set-Up for Tether "Stations" Must be Amortized Over Many Cycles**
- **Current Work :**
 - **STS Demo for Upper Atmosphere Probe**
- **Technology Needs :**
 - **Dynamics / Control**
 - **Materials are SOA Kevlar or Spectra**

EXTRATERRESTRIAL RESOURCE UTILIZATION

Extraterrestrial resource utilization (ETRU) can provide major reductions in IMLEO by producing propellants and other materials (e.g., structures, shielding, etc.) from extraterrestrial resources. ETRU can be used for both lunar and Mars missions. For example, several processes have been investigated for producing oxygen from lunar soil (regolith). One extensively studied concept uses the mineral ilmenite ($FeO{\cdot}TiO_2$), which is about 10% of the lunar regolith, at feedstock. The ilmenite is chemically reduced by hydrogen, producing water, iron, and titanium dioxide. The water is electrolyzed to produce oxygen and hydrogen, and the hydrogen is re-cycled.

On Mars, carbon dioxide, the main component in the Martian atmosphere, is broken down into carbon monoxide and oxygen; the oxygen is then extracted from the gas mixture by means of a zirconia membrane. Zirconia (ZrO_2) has the property of transporting oxygen through its crystal lattice when a voltage is applied across the membrane. This technology is currently under development by the DoE for extraction of oxygen from terrestrial air.

If water is available on the Moon (at the lunar poles, etc.) or Mars (permafrost, polar caps), it can be electrolyzed to produce oxygen-hydrogen propellant at an oxidizer-to-fuel ratio (O/F) of 8. On Mars, water can also be combined with carbon dioxide to produce methane and oxygen at an O/F of 4, which is nearly ideal for propulsion applications. Even if water is not available on Mars, the carbon monoxide produced by extracting oxygen (from carbon dioxide in the Martian atmosphere) can be used as a low-performance (but "free") fuel.

As with several of the advanced concepts discussed above, the benefits of ETRU depend heavily on the infrastructure requirements. These include the materials production facilities, their consumables (e.g., chemical fluxes, electrodes, etc.), and the systems required to transport and set up the "factories". The degree of process closure or recycling is also important, since this impacts the amount of imported consumables required. Also, unmanned precursor missions may be needed to map out resource locations on the Moon (ilmenite, water) and Mars (water).

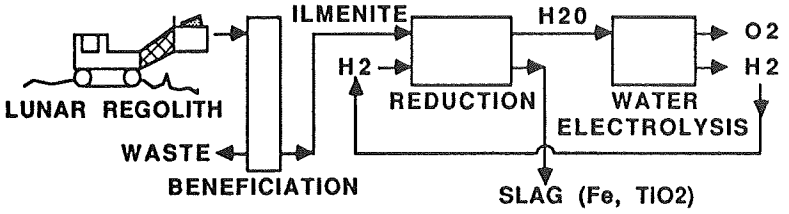
Current work in the area of ETRU processing includes Johnson Space Center (JSC) studies of lunar oxygen production concepts and a University of Arizona project to develop and build a breadboard system to produce oxygen from a simulated Martian atmosphere. Thrusters capable of using propellants derived from ETRU systems (e.g., high-O/F O_2/H_2, O_2/CH_4, O_2/CO, etc.) are under development at LeRC.

Continued thruster development is required to resolve many of the unique technology issues associated with thrusters designed to use unconventional ETRU-produced propellants. These include cooling, coking, and ignition, as well as feed systems, since some potential ETRU-produced fuels are solid materials. For the ETRU production processes, it is necessary to demonstrate the various candidate processes, with the appropriate simulated ET resource, in order to evaluate efficiencies, power requirements, lifetimes, and closure for the various systems. With this information, it will then be possible to select a preferred system for use on the Moon or Mars. For lunar regolith- based systems, it will also be necessary to develop technologies for low- gee soil moving (scooping, digging, etc.) and beneficiation (mineral separation). Finally, ETRU process "factories" will need technologies common to a variety of SEI applications, including power, thermal control (refrigeration), and construction.

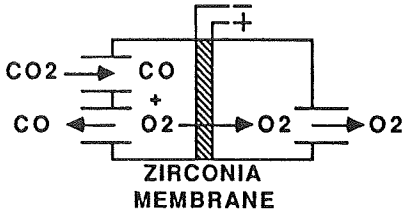
JPL ET RESOURCE UTILIZATION

- **Provides Major Reduction in IMLEO by Producing Propellants, etc., from Local Materials**

Moon : O2 from H2 Reduction of Ilmenite (FeO•TiO2)



Mars : O2 from CO2



- **Mission Benefits Issues :**
 - **Infrastructure ("Factory"), Closure (Imports)**
 - **Precursor Missions to Locate Resources (Ilmenite, Water)**

- **Current Work :**
 - **ET Propellant Thruster Development at LeRC**
 - **JSC-Funded Studies of Lunar O2 Production Concepts**
 - **U of Arizona Breadboard Demo of Mars CO2/O2 System**

- **Technology Needs :**
 - **Demo Thrusters With Unconventional ET Propellants**
 - **Demo Processes w/ Simulated Resource**
 - **Efficiencies, Power, Closure**
 - **For Moon : Demo Digging, Scooping, Beneficiation, etc.**
 - **Related Technologies : Power, Refrigeration, Construction**

SUMMARY

To summarize the advanced propulsion concept discussed in this presentation, MMW class NEP and SEP, solar sails, tethers, and ET resource utilization can have a significant impact on the lunar and Mars SEI. 100 MW_e class NEP is the only near-term concept that appears capable of providing the one-year round-trip time that may be required for a piloted Mars mission. At powers of tens of megawatts, NEP and SEP are both attractive for Mars Cargo missions. The benefit of MMW NEP and SEP is strongly dependent on the power and propulsion system performance; low specific mass is essential and high I_{sp} (up to 10,000 lb_f-s/lb_m) is highly desirable. Finally, development of the NEP vehicle nuclear power system may gain a significant input from the DoE/SDIO MMW space nuclear power program.

Solar sails are the lightest, but also slowest transportation system for cargo missions. Their benefit will be a function of the relative importance of IMLEO as compared to trip time. Interestingly, deployable (Halley's Comet class) sails are potentially the most near-term of the APCs discussed here.

Tethers and extraterrestrial resource utilization can show significant benefits for both lunar and Mars missions. However, their benefit does depend on the infrastructure (tether stations, process factories, initial transportation and set-up) required for their operation. In both cases, the infrastructure must be amortized over many usage cycles. For the ETRU processes, propellants, especially oxygen, will be produced in the near-term. As the technology matures, other materials (structures, etc.) can be produced. However, because of the need to manipulate large quantities of lunar regolith, the technology for lunar oxygen production will tend to be more complex than that for martian oxygen production which requires only atmospheric carbon dioxide as its feedstock.

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SUMMARY

- 100 MWe Class NEP May Enable 1-Year Round-Trip for Piloted Mars Missions
 - NEP and SEP Competitors at < 100 MWe Levels for Cargo Missions or Split Piloted Missions
 - Benefit Depends on Power System and Thruster Performance
 - Low Specific Mass Needed; High Isp Desirable
 - NEP Synergistic with DoE/SDI MMW Space Nuclear Power
- Solar Sails Lightest Transportation System, but also Slowest
 - Benefit Depends on Importance of IMLEO Versus Trip Time
 - Deployable (Halley Comet Class) Sails Very Near Term
- Tethers May Show IMLEO Benefits for Large Missions
 - "Amortize" Infrastructure Over Many Operational Cycles
- ET Resource Utilization Can Provide Propellants in the Near Term and Other Useful Products in the Far Term
 - Lunar Process Complex, Mars Process Simple

CONCLUSIONS

The advanced propulsion concept discussed above can be developed to meet the schedule of SEI missions in the first quarter of the 21st Century. Out of all of these concepts, MMW space power (solar and nuclear) and high- power electric thrusters stand out as two technologies that can significantly enhance and potentially enable a wide variety of piloted SEI missions. However, a major study effort is needed soon to characterize and compare these concepts for SEI missions. This will make it possible to select the best concept(s) and plan the technology development effort required to meet the SEI schedules. Once selected, the total technology development cost for any one concept is likely to be in the $100M to $1B range. Although this is a non-trivial development cost which may be necessary for several technologies, it should be remembered that the savings in IMLEO made possible by this advanced technology will off-set these costs. For example, at a launch cost of $1M per metric ton to LEO (one-tenth the current launch costs), a single Mars cargo mission would save as much in launch costs using an advanced propulsion concept as the costs of developing that concept. Finally, continuous support at the $1M to $5M per year level is essential to maintain basic research on the very advanced concepts which will be required for large-scale solar system exploration and exploration missions in the post-2025 era.

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CONCLUSIONS

- The Advanced Propulsion Concepts Discussed Above Can Be Developed in Time for SEI Missions in the First Quarter of the 21st Century
- MMW Space Power and Advanced High-Power Electric Thrusters Stand Out as Important Technologies That Can Enable a Wide Variety of SEI Missions
- A Major Study Effort Is Needed Soon to Select Among the Concepts and to Plan the Technology Development Effort
- Once Selected, the Total Technology Development Cost for Any One Concept Likely to Be in the $0.1-1B Range
- Support at the $1-5M per Year Level Needed for Basic Research on the Very Advanced Concepts Which Will Be Required for Missions in the Post-2025 Era

FOREIGN TECHNOLOGY

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JAPANESE TECHNOLOGY

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FOREIGN TECHNOLOGY

An Overview of Japanese Space Technology

An assessment of Japan's current capabilities in the areas of space and transatmospheric propulsion is presented. The primary focus is upon Japan's programs in liquid rocket propulsion and in space plane and related transatmospheric areas. Brief reference is also made to their solid rocket programs, as well as to their supersonic airbreathing propulsion efforts that are just getting underway. The results are based upon the findings of a panel of engineers made up of individuals from academia, government and industry, and are derived from a review of a broad array of the open literature, combined with visits to the primary propulsion laboratories and development agencies in Japan. The opportunity to meet with many of the Japanese scientists, engineers and leaders, was the key that made the study possible. Not only did their courtesy and cooperation aid us while we were in Japan, but it also proved to be crucial in helping us establish an identity with research work reported in the literature.

Japan's long term plans for space activity as well as their generic paths for achieving these plans are outlined in the Fundamental Guidelines of Space Policy. This document was originally written in 1978, and has since been revised twice to reflect a rapidly broadening space vision; first in 1984, and most recently in 1989. As with any other such plan, the present version will require continued periodic updating to keep pace with anticipated advances in technology, and changing socio-economic factors. Even a cursory review of Japan's space program shows that it is a very aggressive and forward-looking program. Their on-going activities as well as their planned programs are broad, bold and far-reaching in perspective. Japan's future goals in space include virtually all aspects of space activity.

The current emphasis in Japan's space policy is on developing appropriate internal resources for a variety of space activities. Japan is particularly cognizant of a need to develop an infrastructure for space that will enable them to encourage well-coordinated, but diverse, domestic space development activities, while keeping pace with, and contributing to, international space development. Their motivation arises from a desire to advance basic science and technology, enable expanded participation in international space ventures and satisfy a growing interest in a broad range of domestic space activities. Domestic space interests encompass activities that exploit the unique environmental conditions of space, prepare for civil space development, and promote manned space activities. Japan's plans for international collaborations include cooperation in programs established by other countries, initiation of collaborative programs of their own, including

regional cooperative projects in the Asian Pacific, and assisting developing countries with space activities.

Japan's space program is founded upon two basic tenets which underscore all their activities. The first is that they wish to develop "assured access" to space, while the second is that their space activities are for "solely peaceful" purposes. Although they encourage international cooperation in space, concerns may sometimes arise in conjunction with potential collaborations with the US, because synergisms between NASA and Air Force programs may conflict with their guideline for purely peaceful uses of space. Further, their "assured access" policy dictates that they develop autonomous capabilities in space, which in many instances will duplicate capabilities in other countries. Japan's space aspirations, however, leave ample room for cooperative Japan-US space endeavors and it was clear from our visit that they are committed to establishing new joint ventures with the US as well as continuing existing ones. Future joint ventures between Japan and the US would appear to be mutually beneficial.

Japan's goals for the present decade include plans for continuing their already strong thrust in scientific space research, for bringing their satellite and launch technologies up to those of international standards, for creating the infrastructure for Space Station activities, and for developing the basic technologies required for their own manned space activities. These near-term goals include the promotion of advanced satellite technologies such as their Engineering Test Satellite (ETS) series and communication, broadcasting and navigation satellites, culminating in the development and manufacture of commercial satellites. Importance is also given to the development of scientific satellites with supporting efforts in space sciences, facilities, and tracking and control systems. These scientific areas are seen as being particularly appropriate for international cooperation. Japan's near-term plans also reaffirm their significant participation in the US Space Station through the Japan Experiment Module (JEM) and SSIP modules. These collaborative efforts will serve to develop basic domestic technologies. Eventually Japan's plans call for an independent manned spacecraft, built on current technology programs.

An area of primary emphasis in Japan's near-term space plans, and one that is also of central focus in this report, is the establishment of their own space transportation system. Self-assured access to space is envisioned as being indispensable to Japan's long-range space development activities. Primary near-term goals in space transportation are the development of an expendable launch system for transportation of materials to geostationary orbit, the establishment of a technology for unmanned space to ground

transportation and the promotion of fundamental research and development for long-term manned space transportation capabilities. Current transportation plans for expendable launch vehicles are focussed on developing and enhancing the H- and M-series of liquid and solid rocket systems. The H-series liquid rocket system, which will ultimately provide commercial launch capabilities for Japan, are addressed in the first half of the present report.

Japan's long-term goals for the first decade of the new century and beyond include the implementation of their own manned space capabilities, the launch and operation of a geostationary platform, the development of an orbital servicing vehicle and an orbital transfer vehicle, and the ultimate development of their own space station. Japan also places much emphasis on the commercial uses of space with plans for manufacturing experiments, materials development and a space factory. The advanced transportation capabilities required for these activities are discussed in detail in the spaceplane and transatmospheric propulsion sections of the present report.

The space program in Japan is under the auspices of the Space Activities Commision (SAC), a cabinet level body that oversees the space activities of the entire country. The primary operative body under SAC is the Space Technology Agency (STA) which oversees and coordinates the efforts of all space programs in Japan. There are three primary agencies devoted to space initiatives. These are the Institute of Space and Astronautical Science (ISAS), the National Aerospace Laboratory (NAL) and the National Space Development Agency of Japan (NASDA). Each of these agencies has major responsibility for certain areas of space initiatives.

Current budget levels (FY 1989) for Japan's space program including satellites, launch vehicles and propulsion systems is about 176 billion yen ($1.26B US) plus additional private funding. Detailed plans of the various agencies and laboratories as well as the expanded goals set forth in the 1989 update of the Foundations of Space Policy document, suggest that this amount will grow substantially in the future. Specific plans are in place to strengthen the Space Activities Commision, to increase the breadth and depth of technical staff in related research and development institutions and to add faculty and upgrade equipment in universities and encourage academic institutions to engage in space-related research and development activities. To keep pace with these plans, annual space program growth rates in excess of 10% are forecast for the foreseeable future. Guidelines for space budgets are targeted at a level commensurate with Japan's current 10% share of the world economy. Both government and the private sector will be called upon to share in funding this increased space activity. To

strengthen and encourage private sector participation, the government will promote financing strategies, tax incentives and other considerations including provisions for enabling the private sector to participate in various space activities at reduced costs.

There are several major space transportation efforts in Japan including three expendable rocket launch vehicle programs and three airbreathing hypersonic vehicle concepts. The rocket launch vehicles include both operational systems and ones under development, the N-series, the H-series and the M-series, while all the airbreathing hypersonic vehicles are in the concept definition phase. The N-series of launch vehicles was based upon US technology developed under license, while the currently operational H-I vehicle includes technology that is partly based on Japanese design and development and in part retains technology developed under license from the US. The H-II vehicle, which is currently under development and scheduled for first use in 1993, is completely Japanese in design and positions Japan as a full-fledged member of the world launch community. The M-series rockets are solid boosters that have long been based upon Japanese design.

Japan's launch facilities at Tanegashima are located at 30.4 N lattitude, a location that is nearly the same as our launch facilities at Kennedy Space Center, which are at 28.5 N. The size of the launch site is much smaller than KSC, and the transportation facilities in the immediate area are somewhat limited, but they appear to be adequate for the H-II. A current agreement with local residents limits launch windows to a few weeks per year, but plans for a public education campaign to inform local residents and interest groups of the importance of launch functions to Japan's national needs are in progress. This nationwide campaign will seek to encourage understanding of Japan's space development activities and to foster an environment conducive to space development.

The propulsion sources for Japan's various transportation efforts encompass some eight major development programs which serve as the focal point for most of the present report. These programs are in various stages ranging from concept development to operational. They include four cryogenic hydrogen-oxygen rocket engines and four advanced airbreathing systems which are not as far along in development as the rocket engines.

In conjunction with current H-series expendable launch vehicle programs, propulsion development is on-going for the LE-5a, and the LE-7 cryogenic propulsion engines. The HIPEX, expander cycle engine, represents an additional new major liquid hydrogen-oxygen engine development that is

currently underway. The LACE liquid air cycle engine which is also in advanced development is a generic propulsion system oriented towards advanced airbreathing systems such as strap-on boosters for up-rated versions of the H-II or hypersonic propulsion applications. The sixth engine configuration is the ATREX engine, an air turboramjet system which is in a similar development stage. The remaining two propulsion systems are a SCRAMJET engine concept development program at NAL for eventual hypersonic applications, and the newly announced Mach 5 turbojet/turboramjet engine development which is being supported by MITI for high speed commercial transportation in the Pacific rim area.

The systems and performance of Japan's cryogenic liquid rocket engines are comparable to that of engines developed in the United States. In their designs, they have made extensive use of US data, procedures and technology and their engines have similar specific impulse and vacuum thrust to weight ratios. The new engines are; however, decidedly their designs, and show a number of significant differences from US systems. Their engine development programs, which are built upon a phased project management concept similar to that used by NASA and USAF, are composed of carefully planned steps involving low risk, well-characterized options, allow necessary adjustment of engine designs as the experimental results dictate. The general result is a conservative design that is heavily based upon experimental engine testing. The slightly more conservative design should facilitate reliablility, and may be particularly beneficial when these engines and/or their derivatives are man-rated.

In most of Japan's space propulsion program, the emphasis is based upon building a launch capability to fill a need. The design requirements are set by the end product's use. Whereas, for example, the need for man-rated reliability, reusability and high performance has driven turbopump designs for the SSME, the Japanese have placed emphasis on expendable launch vehicles with low cost and limited life. They will undoubtedly delay man rating their engines until their engine developments have become more mature. Manned activities emphasize longer life, improved diagnostic measurements, and, in general, a well-perfected product, areas in which Japan has long demonstrated expertise. Certainly Japan's capabilities in fabrication and manufacturing as well as their broad based expertise in high technology in general, place them in a position to make very rapid advances in space capabilities and to contribute effectively to the world's space activities.

In the area of turbomachinery, the Japanese turbopumps and turbines again demonstrate performance levels that are similar to US capabilities. The

Japanese are behind the US in some areas of turbomachinery, but they are ahead in others. Their basic approach to design is to first ascertain the technology level, then apply an adequate margin to increase the probability of success, conduct component testing to verify and anchor the design, and then to proceed with the flight version. For example, in one instance, they have chosen a two stage over a three stage pump to avoid a technology development program. Their overall effort is a cooperative one that minimizes duplication of effort and maximizes the rate of advancement.

In the transatmospheric and hypersonic propulsion area, the Japanese are beginning a study of space plane concepts that emphasizes diverse topics such as aerodynamics, structures, slush hydrogen fuel, CFD, advanced propulsion and system development scenarios. The propulsion cycles under study are similar to those being considered in the US, and include the turbojet, the ramjet, the turboramjet or and the supersonic combustion ramjet (SCRAMJET). The propulsion systems of primary interest appear to be those for the Mach 3 to 6 range for the low Mach number portion of hypersonic cruise or SSTO vehicles, strap-on booster augmentation engines for launch systems, or airbreathing engines for a civilian SST. Their efforts in higher Mach number propulsion systems are directed more toward accumulating a data base.

In terms of engine development, there are two classes of engine that are presently in the prototype phase; the LACE engine at MHI, and the ATREX, air turboramjet, at IHI. There was also reference to the development of a turboramjet engine at KHI, but even though this is probably the least complex and risky cycle, it does not appear that engine components are presently available for this engine. Demonstration engines are currently available (or nearly so) for the LACE and ATREX engines, but the development programs have been put on temporary hold because all LH2 facilities are now dedicated to the LE-7 development effort.

The LACE demonstrator engine uses the LH2 pump and combustor from the LE-5 engine, along with new components for the air liquefier and the liquid air pump. This adaptation of components from existing rocket programs to new propulsion efforts is chacteristic of Japanese space propulsion programs. They do a very effective job of using previously demonstrated components in advanced projects. In addition to the LAC E engine, the HIPEX and the ATREX engines also contain heavy commonality with the liquid rocket engines. The ATREX engine relies upon IHI's existing turbojet-turbofan production and design experience as well as the expander cycle technology developed in the HIPEX engine. This interchangeable component technology appears to be providing very cost-effective progress

in Japan's new programs, while simultaneously enhancing the reliability of their liquid engines as well.

Although a considerable amount of technology development is directed toward SCRAMJET applications, the Japanese program in this area is only in the concept definition phase, and demonstration engine development does not appear imminent. Japan appears to have significant interest in the development of a hypersonic vehicle as a member of a consortium, instead of all alone. The general feeling is that the technology is now available for the LACE and ATREX engines, but that technology for the SCRAM engine is not yet accessible.

The SCRAMJETtechnology programs include considerable emphasis on experimental studies of supersonic combustion including ignition and diffusion flame studies and shock tube studies of elementary reaction kinetics of hydrogen. In addition, high speed inlet tests are currently underway on a scale model. This work takes place in the national laboratories and at several universities. Two new university efforts that involve some 20 faculty at several schools and are oriented towards hypersonic reacting flows and component technology for advanced propulsion systems are also underway. To complement these experimental studies, CFD studies of SCRAMJETconfiguations are being conducted at NAL Chofu where they are using this experimental data to validate and anchor their CFD codes.

In terms of facilities, there are SCRAMJET facilities at NAL Chofu, NAL Kakuda and the University of Tokyo which all have capabilities for Mach 2. A new SCRAMJET facility is also being built at Kakuda. The Japanese also plan to construct an engine test facility at Kakuda for testing supersonic airbreathing engines. This will be a key facility in MITI's recently announced engine development program for a high speed civil transport.

Japan also is placing attention on advanced fuels development and on plant construction for hydrogen production. Japan has developed two high density hydrocarbon fuels for rocket applications, and is in the process of stepping up their hydrogen production capabilities to serve the H-II and advanced airbreathing propulsion propulsion systems. They are currently constructing a new hydrogen plant that makes hydrogen as the byproduct of ethylene production, and are building a pilot facility for the production of hydrogen from coal gasification.

In the area of advanced diagnostics, Japan is a user of the latest systems from the US and Europe, but are leaders in the development and manufacture of many of the basic lasers, optics and electooptic components that go into these systems. Of particular interest to advanced diagnostics implementations are new tunable diode lasers that are being developed in Japan and a new surface emitting diode laser with reduced beam divergence that offers possibilities for higher spatial resolution.

The area of computational fluid dynamics (CFD), which is an important supporting area in all propulsion development, represents an area of strength in Japan. Their domestic supercomputers are among the best in the world, and they have major supercomputer installations at NAL and at the privately owned Institute for Computational Fluid Dynamics. The national universities also have excellent supercomputing capabilities. This abundance of supercomputer access has resulted in rapid progress in computational areas. The Japanese routinely include real gas effects and complex reaction kinetics in flowfield analyses, and their codes are based on the latest algorithms. Their visualization and postprocessing capabilities are also at the leading edge. Clearly they have appropriate CFD capabilities to enable them to move rapidly in this aspect of propulsion development.

Finally, we note that contractor selection in Japan is an area in which there are differences from that in the US. Although competition exists, particularly at the concept development level, the award of new propulsion contracts generally based on the technical capabilities which the contractors have demonstrated in previous projects. For example, MHI is generally the overall engine developer for liquid rocket engines, while IHI will generally emerge as the turbomachinery contractor. The project share generally appears to be set by historical factors, rather than by competitive procedures.

Their industry role is coordinated and strengthened through the Keidanren and the Society of Japanese Aerospace Companies (SJAC).

The LACE cycle is effective up to flight Mach numbers of 6 to 8. Both HIPEX and the LACE systems are in advanced development states with engine hardware currently available for near term test.

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ME/PERC
Penn State University

A REVIEW OF LIQUID ROCKET PROPULSION PROGRAMS IN JAPAN

Charles L. Merkle
Penn State Propulsion Transportation Symposium
June 25-29, 1990
The Pennsylvania State University

JAPAN'S SPACE PROGRAM: NATIONAL POLICY

- FUNDAMENTAL GUIDELINES OF SPACE POLICY
 - Long-Term Plans
 - Generic Paths for Achieving Plans
 - Issued 1978
 - Revised 1984, 1989
- CURRENT EMPHASIS IN SPACE POLICY
 - Developing Internal Resources for Far-Reaching Space Program
 - Diverse Domestic Space Activities
 - Contributors to International Space Development
- BASIC TENETS
 - "Assured Access"
 - "Solely Peaceful Purposes"

JAPAN'S SPACE PROGRAM: GOALS AND PLANS

- DOMESTIC SPACE INTERESTS
 - Prepare for Civil Space Development
 - Promote Manned Space Activities
 - Exploit Environmental Conditions of Space
- INTERNATIONAL COLLABORATIONS
 - Cooperate in Programs Established by Other Countries
 - Initiate Collaborative Programs of their Own
 - Regional Projects in Asian Pacific
 - Assisting Developing Countries

Japan's Space Program

- Expendable Launch Vehicles
- Communications and Broadcast Satellites
- Weather Satellites
- Earth Observation Satellites
- Robotics Systems
- Space Station Components
- Manned Vehicles

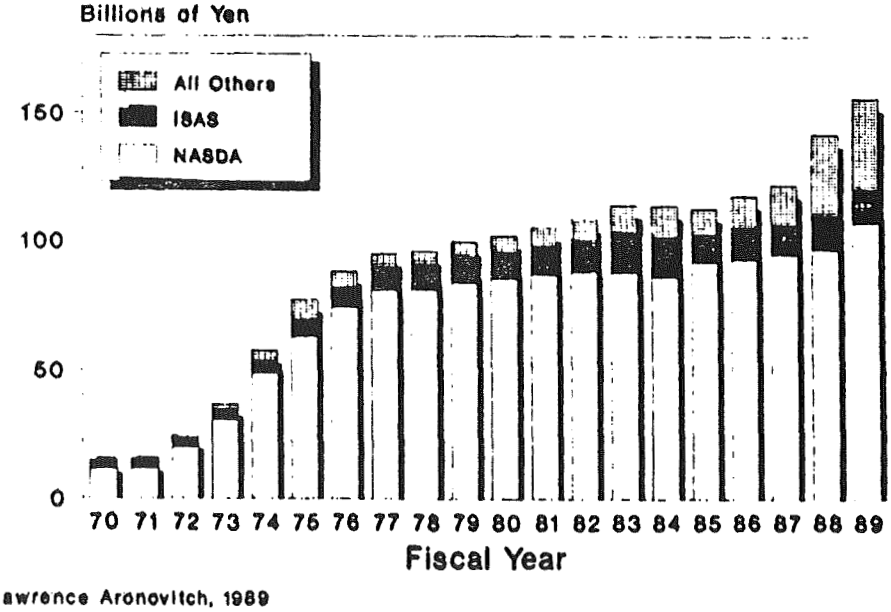
JAPAN'S SPACE PROGRAM: NATIONAL ORGANIZATION

- SPACE ACTIVITIES COMMISSION
 - Oversees All Space Activities
 - Cabinet Level
- SPACE TECHNOLOGY AGENCY
 - Primary Operative Body
 - Coordinates All Space Programs
 - Provides Major Funding
- ADDITIONAL EFFORTS
 - Ministry of Education (Mombusho)
 - Ministry of International Trade and Industry (MITI)
- THREE PRIMARY R&D AGENCIES
 - Institute of Space and Astronautical Science (ISAS)
 - National Aerospace Laboratory (NAL)
 - National Space Development Agency (NASDA)

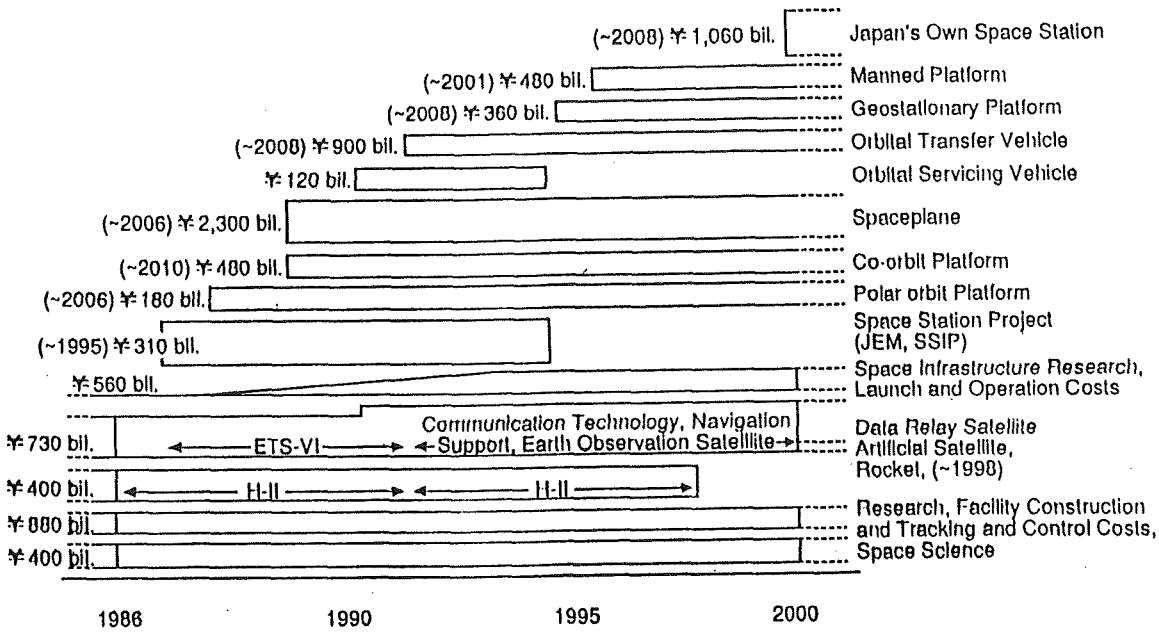
NATIONAL ORGANIZATION FOR SPACE

ORGANIZATION	CHARTER	PERSONNEL	BUDGET
NAL	Aeronautical and Space Technology	450	¥ 10,000 M $ 70 M
NASDA	Applications Satellites and Launch Vehicles	950	¥ 110,000 M $ 840 M
ISAS	Scientific Satellites and Launch Vehicles	291	¥ 21,000 M $ 150 M
Others			¥ 27,000 M $ 194 M
TOTAL			¥ 176,000 M $ 1,254 M

JAPANESE SPACE BUDGETS
(By Agency)



Lawrence Aronovitch, 1989



Major Projects

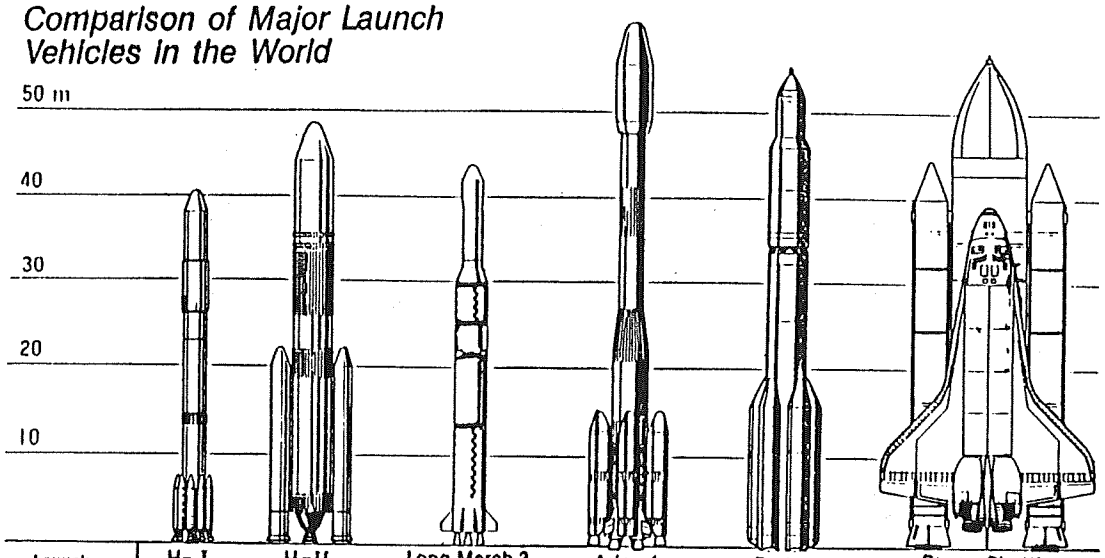
INSTITUTE OF SPACE AND ASTRONAUTICAL SCIENCE (ISAS)

- National Interuniversity Research Institute
 - *Ministry of Education (MOMBUSHO)*

- Objectives: Research in Space Science
 - Scientific Satellites
 - Launch Vehicles
 - Sounding rockets
 - Balloons

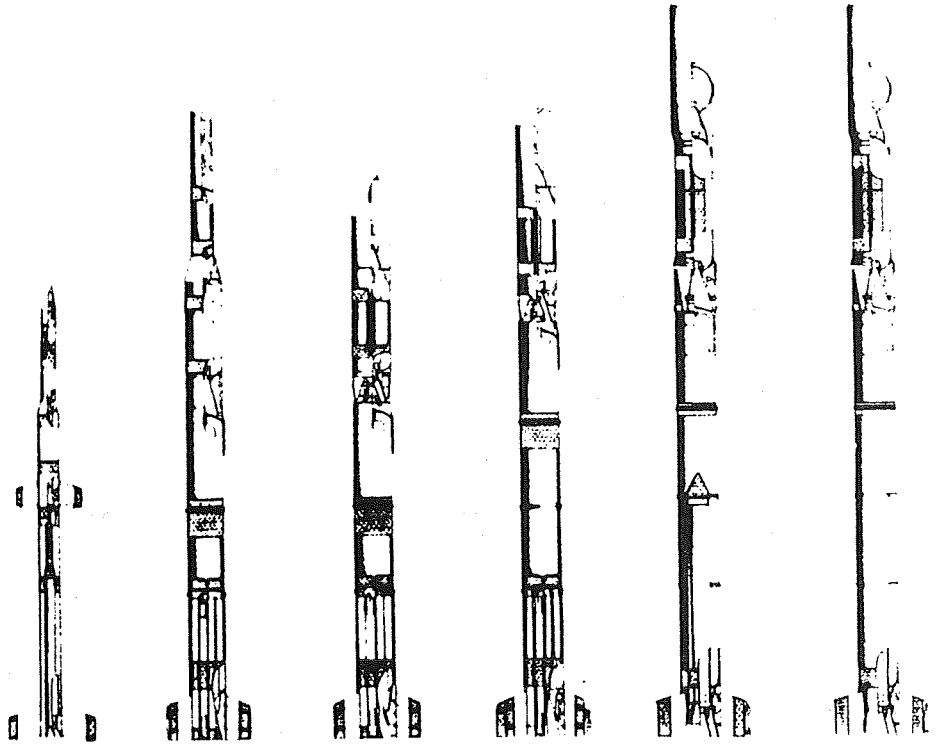
- Launch Vehicles - M Family

Comparison of Major Launch Vehicles in the World



Launch Vehicles	H-I Japan	H-II	Long March-3 China	Arlan-4 Europe	Proton U.S.S.R.	Space Shuttle U.S.A.
Gross Weight	140 t	260 t	202 t	460 t	680 t	2,041 t
Launching capability into a geostationary orbit	550kg	2,200kg	650kg	2,200kg	2,000kg	2,270kg (Using upper-stage)

ISAS Launch Vehicles



	L-4S	M-4S	M-3C	M-3S(M-3H)	M-3SII	Improved M-3SII
Number of Stages	4	4	3	3	3	3
Total Length	16.5 m	23.6 m	20.2 m	1.41 m	27.8 m	30 m
Diameter	0.735 m Ø	1.41 m O	1.41 m O	0.735 m O	1.41 m O	2.5 m Ø
Total Weight	9.4 ton	43.6 ton	41.6 ton	48.7 ton	61 ton	120 ton
Payload*	Approx 26kg	Approx 180kg	Approx 195kg	Approx 290kg	Approx 770kg	Approx 2000kg

*Orbiting capability onto the circular orbit of 250km height with 31° inclination.

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H-II LAUNCH VEHICLE

- Two-Stage Rocket with Two Strap-on Solid Boosters
- Booster Engines
 - 14% HTPB / 18% Al / 68% AP
 - 4 Segments/Booster
- First Stage -- LE-7
 - LOX/LH_2
 - Staged Combustion
- Second Stage -- LE-5A
 - $LOX\text{-}LH_2$
 - Expander Bleed Cycle

NATIONAL AEROSPACE LABORATORY (NAL) OVERVIEW

OBJECTIVES: Aeronautical and Space Technology

- **Budget:** $70 M
 - Personnel $20 M
 - Research $30 M
 - Facilities $20 M
- Budget Flat Since 1982
 - Near-term Growth Expected
 - Major Increase in Facilities Budget Since 1982
- **Personnel:** 450
 - 325 Research
 - 125 Other.

MAJOR PROJECTS:

- ASKA STOL Aircraft 1977-88
 - Design, Manufacture, Flight Test
- Innovative Aerospace Technologies, 1987 -
- LE-7 LOX Turbopump Initial Development

NATIONAL SPACE DEVELOPMENT AGENCY (NASDA) OVERVIEW

- Objectives: Applications Satellites and Launch Vehicles
- Budget: $840 M
 - Approximately Flat Since 1982
 - Near-term Growth Expected
- Personnel: 950
- Major Projects:
 - Launch Vehicles
 - Satellites
 - First Material Processing Test (Shuttle/Spacelab)
 - Japanese Experiment Module (JEM)

LE-5A Engine for H-2 Second Stage

Expander Bleed Cycle
Propellant - LOX/LH_2

Thrust = 26,460 Lbf

Mixture Ratio = 5.0

Specific Impulse = 452 sec

Chamber Pressure = 570 PSIA

Nozzle Area Ratio = 130

Burn Time = 525 sec

Weight = 540 Lb

Status - Qualification

LE-7 Engine For H-2 First Stage

Staged Combustion Cycle

Propellant - LOX/LH_2

Thrust = 265,000 lbf, vac

Mixture Ratio = 6.0

Specific Impulse = 451 sec

Chamber Pressure = 2133 psia

Nozzle Area Ratio = 60

Burn Time = 315 sec

Weight = 3439 lb

Status - Development

UPPER STAGE ENGINE TURBOMACHINERY

LE-5 vs RL10

	Hydrogen		Oxygen	
	LE-5	RL10A-3-3A	LE-5	RL10A-3-3A
Number of stages	1	2	1	1
Mass flow rate lb/sec	7.8	6.2	42.7	30.8
Speed, RPM	50,000	32,800	16,500	13,100
Pump discharge pressure, psia	825	1120	740	670
Impeller tip speed ft/sec	1250	1010	315	250
Shroud impeller	Yes	No	Yes	Yes
Efficiency, %	59	57	66	63

BOOSTER STAGE ENGINE TURBOMACHINERY

LE-7 Vs SSME - Hydrogen

	LE-7	SSME
Pump		
Number of stages	2	3
Mass flow rate,lb/sec	87	149
Speed, RPM	46,100	34,100
Pressure rise, psi	4,700	5,800
Efficiency, %	71	77
Turbine		
Number of stages	1	2
Inlet pressure,psia	3,520	4,920
Inlet temperature, °R	1,770	1,780
Pressure ratio	1.43	1.45
Efficiency, %	72	82

BOOSTER STAGE TURBOMACHINERY

LE-7 Vs SSME - Oxygen

	LE-7	SSME
Main Pump		
Number of stages	1	1
Mass flow rate,lb/sec	505	1,070
Speed, RPM	20,000	27,200
Pressure rise,psi	3,030	3,730
Efficiency, %	75	67
Preburner Pump		
Mass flow rate,lb/sec	96	100
Pressure rise,psi	1,650	3,029
Efficiency, %	65	80
Turbine		
Inlet pressure, psia	3,410	4,984
Inlet temperature ^{o}R	1,750	1,455
Pressure % ratio	1.43	1.51
Efficiency	49	79

UPRATED H-II VEHICLES

- Concepts Currently Under Study
 - Increase Number of Solid Boosters
 - Replace SRB's with Liquid Propellant Boosters
 - LOX/CH_4
 - LOX/RP
 - LOX/LH_2
- Payload Increase from 2.2 to 5.5 tons

SUMMARY

- o Agressive Space Policy
 - Broad-Ranging Focus
 - Self-Assured Access to Space
 - Plan Major Role in Space Development/Exploration
- o Emphasis to Date on Expendable Launch Vehicles
 - Current Efforts on Airbreathing Propulsion
 - Future Plans for Manned Flights
- o Government Organizations Significantly Smaller than U.S.
 - NASDA 1000 people
 - NAL 500
 - ISAS 300

SUMMARY (CONT'D)

- o Basic Lauch Vehicles:
 - H-I operational
 - H-II 1993
 - Uprated H-II versions planned
- o Current Rocket Engine Focus
 - LE-5 LE-5A
 - LE-7 HYPEX
 - Have Demonstrated Thrust Levels from 2000 to 260,000 lbs.
 - Engine Performance Generally on Par with U.S. Engines

PRESENTATION 1.5.2

RUSSIAN TECHNOLOGY

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N91-28220

Space Transportation Propulsion
USSR Launcher Technology — 1990

June 1990

Rocketdyne — Advanced Programs
Special Programs Office
R. Jones, Program Manager

Rockwell International
Rocketdyne Division

AGENDA

- ENERGIA Background
 - Launch Vehicle Summary
 - Soviet Launcher Family
 - ENERGIA Propulsion Characteristics
- ENERGIA Propulsion Characteristics
 - Booster Propulsion
 - Core Propulsion
 - Growth Capability

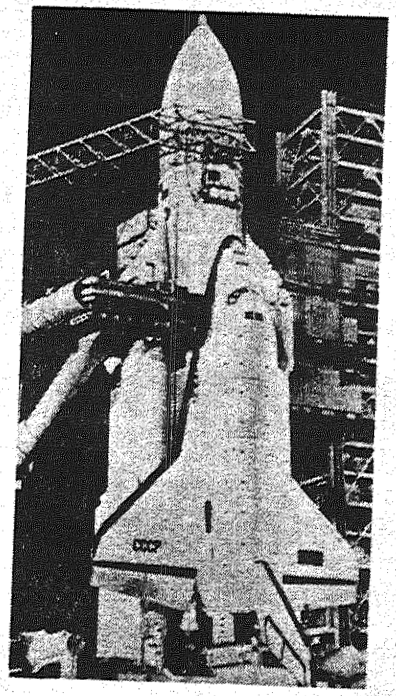


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United States and Soviet Union STS Operations



STS-26 America's Pride:
The Journey Continues



ENERGIA on Launch
Pad with Buran (Blizzard)
Side-Mounted

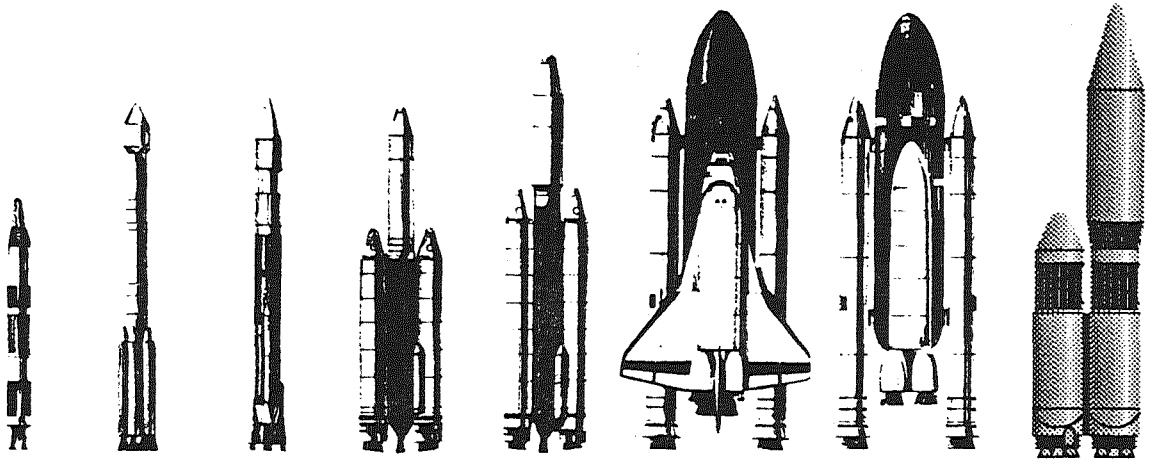
- Buran initial flight achieved 15 November 1988 as United States STS returns to operation (STS-26, Discovery) 29 September 1988
- United States STS operations continue:
 - STS-27, Atlantis, 2 December 1988
 - STS-29, Discovery, 13 March 1989
 - STS-30, Atlantis, 4 May 1989
 - STS-28, Columbia, 8 August 1989
 - STS-34, Atlantis, 18 October 1989
 - STS-33, Discovery, 22 November 1989
 - STS-32, Columbia, 9 January 1990
 - STS-36, Atlantis, 28 February 1990
 - STS-31, Discovery, 24 April 1990
- United States next flight:
 - STS-38, Atlantis, Summer 1990 (secret mission)
 - STS-35, Columbia, Summer 1990 (Astronomical observatory, UV & X-ray telescopes – 1 week)
- Soviet Union next flight:
 - Late 1991 – unmanned of several days duration, docking with MIR, equipment checkout & rescue simulation, automatic return

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U.S. Launch Vehicles

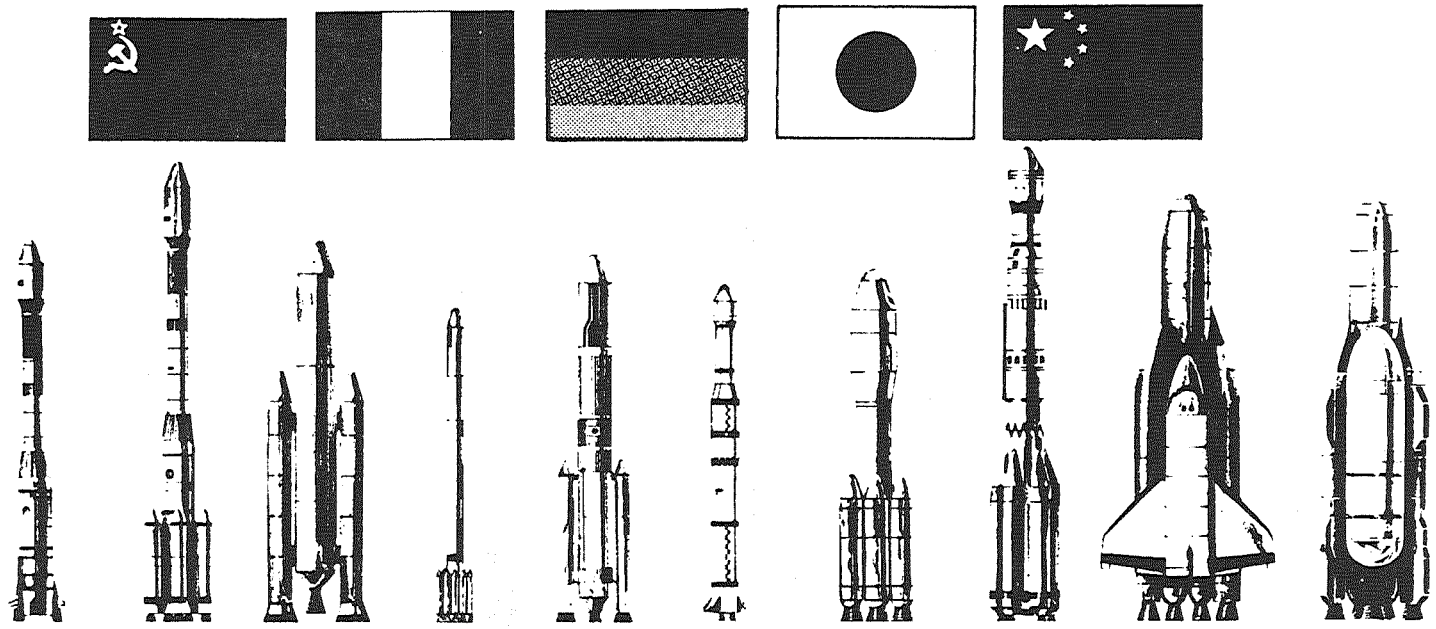


Class	Titan 2	Delta 2	Atlas Centaur	Titan 3	Titan 4	Shuttle	Shuttle C	ALS
LEO (lb)	5,500	11,100	12,300	33,000	42,000	55,000	100,000 -150,000	80,000-220,000
GEO-Transfer (lb)	–	3,190	5,200	8,600	12,500	–	–	–
GEO-Circular (lb)	–	1,350 (PAM-D)	2,500	4,200 (IUS)	10,000 (Centaur)	1,350 (PAM-D) 5,000 (IUS) 6,500 (TOS)	20,000 (Centaur)	–
Inventory	13 Being Refurbished*	20 Ordered	18 Ordered	Reservations for 19 Satellites	48 Ordered	–	–	–
Flight Schedule	First Flight Mid 1988	First Flight Late 1988	First Flight 1990	First Flight 1989	First Flight Late 1988	Return-To-Flight 1988	First Flight (4 years from go-ahead)	First Flight 2000

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International Launch Vehicles

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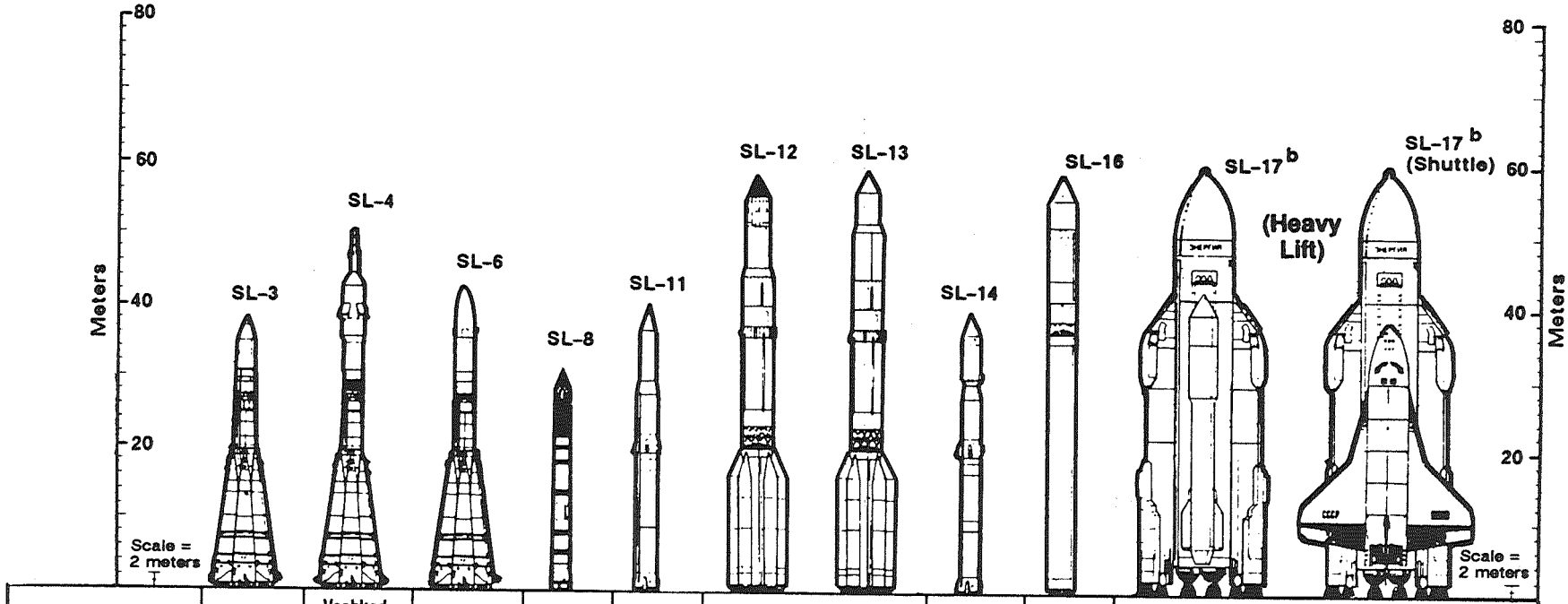


Class	Ariane 3 (Arianespace)*	Ariane 4 (Arianespace)*	Ariane 5 (France)	H-1 (Japan)	H-2 (Japan)	Long March-3 (PRC)	Long March-4 (PRC)	Proton (CCCP)	Energia with Orbiter (CCCP)	Energia HLLV (CCCP)
LEO (lb)	—	—	46,200 (Hermes Spaceplane)	—	—	—	20,000	44,000	66,000	220,000
GEO-Transfer (lb)	5,690	9,260	14,960	2,300	9,000	5,500	8,800	10,000	—	—
LEO-Circular (lb)	2,780	4,800**	7,750	1,210	4,000	2,500	4,000	4,800	—	—
Advertised Flight Rate	4/year	4/year	—	2/year	2-4/year	2-5/year	—	—	—	—
Flight History	17/21 (81%) (Ariane 1, 2 & 3)	First Flight Mid 1988	First Flight 1995	Initial Two Flights 1986/1987	First Flight 1992	Initial Two Flights 1987	First Flight 1991	122/132 (92.4%)	First Flight Nov 1988	One Test Flight May 1987

* European consortium dominated by France (60%) and West Germany (20%)
** Six version of Ariane 4 ranging from 2,300 to 4,800 lbs GEO-Circular

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USSR Launcher Family



	Vostok	Voshkod-Soyuz	Molniya	Cosmos		Proton		Tsyklon	Zenit	ENERGIA	
Lift-Off Weight (kg)[a]	290,00	310,000	310,000	120,000	180,000	680,000	670,000	190,000	400,000	2,300,000	2,300,000
Lift-Off Thrust (kg)[a]	410,00	420,000	420,000	160,000	280,000	900,000	980,000	280,000	600,000	3,550,000	3,550,000
Payload to 180 km (kg)[a]	6,300	7,500	–	1,700	4,000	–	21,000	5,500	15,000+	105,000	30,000

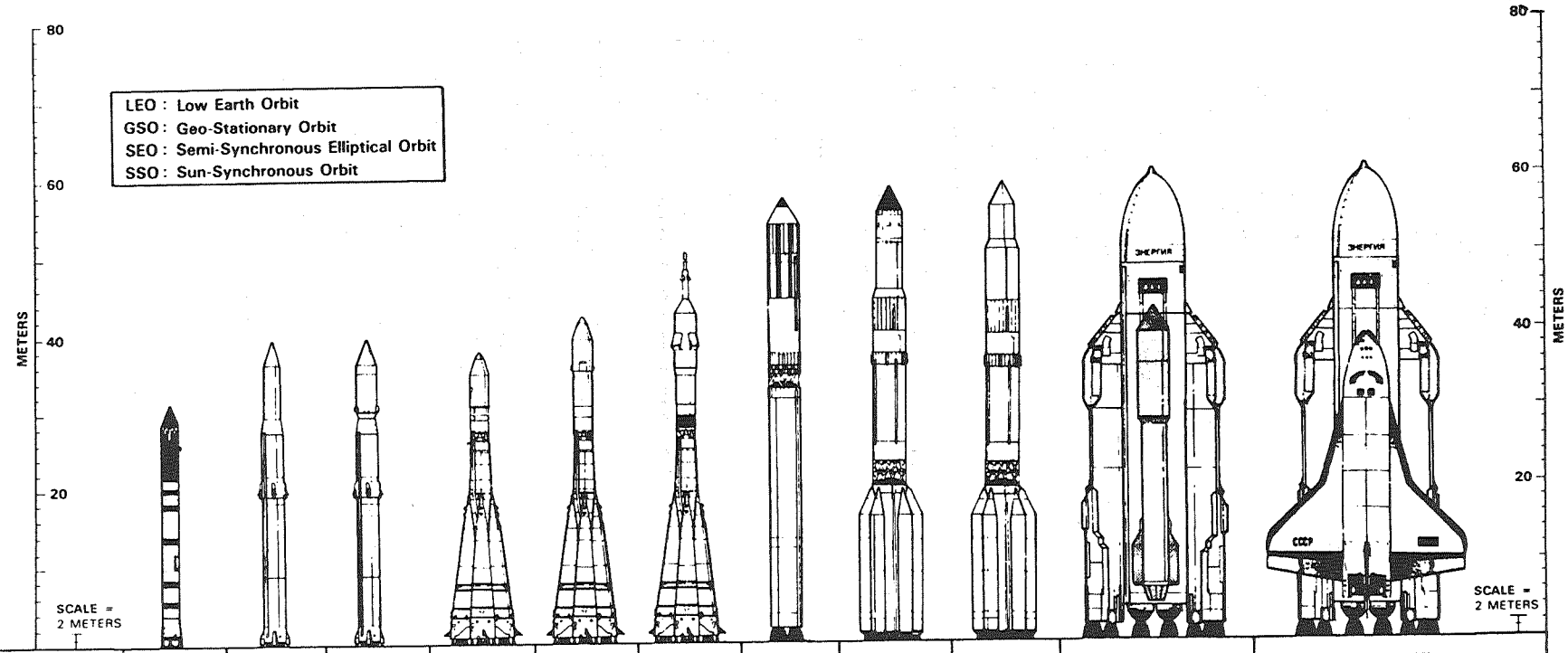
[a]Approximate

[b]In final stages of development

- **With the successful launch of the Buran shuttle, the Soviet launch vehicle arsenal now includes 11 major variants capable of launching payloads from 1,700 to more than 100,000 kg**

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LAUNCH VEHICLES



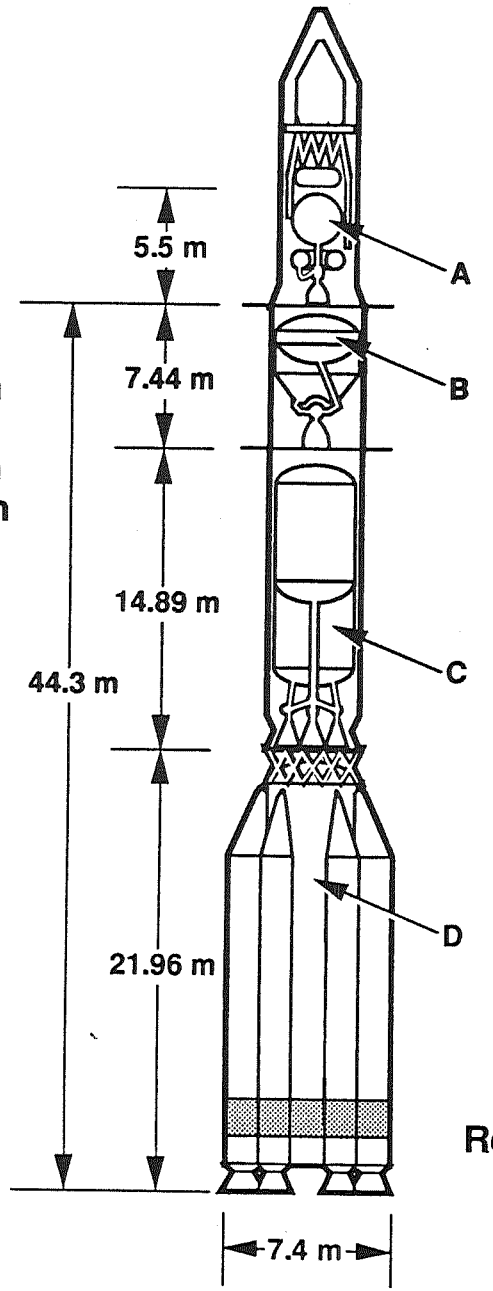
USSR NAME	KOSMOS	—	TSYKLON	VOSTOK	MOLNIYA	SOYUZ	ZENIT	PROTON	PROTON	ENERGIYA	ENERGIYA/BURAN
US DESIGNATOR	SL-8	SL-11	SL-14	SL-3	SL-6	SL-4	SL-16	SL-12	SL-13	SL-17	SL-17
AVERAGE ANNUAL LAUNCHES (1980's)	15	5	7	2	11	41	2[a]	9	1	1[b]	1[c]
OPERATIONAL PAYLOAD CAPACITY (KG)	1,350(LEO)	—	4,000(LEO)	4,730(LEO) 1,840(SSO)	1,800(SEO)	7,240(LEO)	13,740(LEO) 11,380(SSO)	5,700(Moon) 5,300(Venus) 4,600(Mars) 2,500(GSO)	20,600(LEO)	105,000(LEO) 32,000(Moon) 28,000(Mars/Venus) 19,000(GSO)	30,000(LEO)

[a]SINCE 1985 [b]SINCE 1987 [c]SINCE 1988

TELEDYNE BROWN ENGINEERING

Soviet
Proton Space Launcher
SL-12

- Payloads (t)
 - 20.0 - 200 km-Earth orbit
 - 5.7 - Lunar mission
 - 5.3 - Venus mission
 - 4.6 - Mars mission
 - 2.2 Geostationary mission



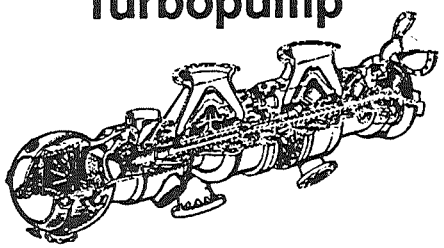
- Fourth Stage (A)
 - One engine (staged combustion)
 Thrust = 84.35 kN (18,963 lb)
 - LOX/kerosene
- Third Stage (B)
 - One engine same as the 2nd stage
 Thrust = 0.6 MN (134,885 lb)
 - Four chamber (open cycle) for thrust vector control with a thrust = 30 kN (6,744 lb)
- Second Stage (C)
 - Four engines (staged combustion)
 Thrust = 4 x 0.6 MN (539,542 lb)
 - N_2O_4 /UDMH
- First Stage (D)
 - Six RD-253 engines
 Thrust = 6 x 1.474 MN (1,988,210 lb)
 - N_2O_4 /UDMH

Ref: Aviation Week & Space Technology, October 13, 1986, page 19
Encyclopedia of Cosmonautics, page 307

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Rocketdyne Division

RD-253 Engine Schematic

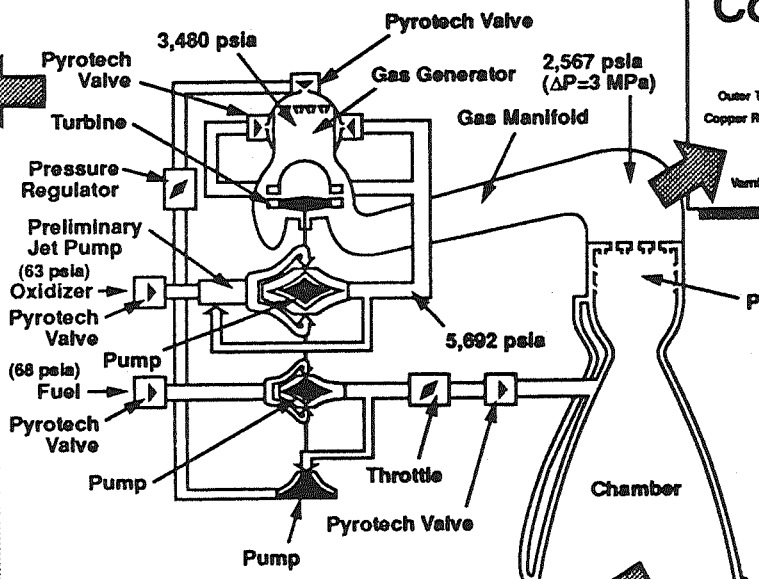
Turbopump



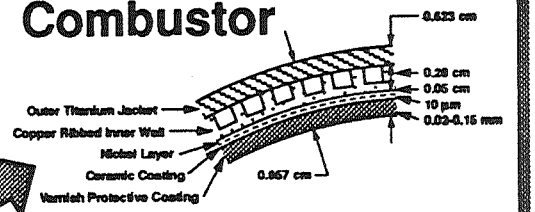
RD-253 Version 1 - Design Concept

1 - Pump (UDMH)
2 - Pump (N_2O_4)
3 - Preburner
4 - Mounting flange (aux. pump spin cartridge)
5 - Axial-flow turbine
6 - Spin turbine

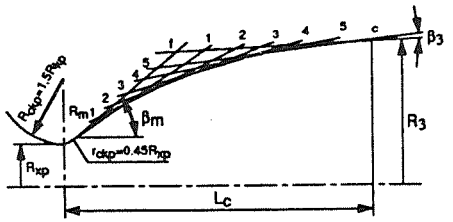
- Single shaft with three centrifugal impellers with double entry & axial-flow reaction turbine
- Maximum N_2O_4 pump discharge pressure, 5,692 psia
- Maximum rotational speed, rpm = 13,860
- Weight = 551 lb
- % of engine weight = 20
- Fuel, UDMH, pump is 2-stage
- Turbine output, 25,130 hp
- Pressure ratio = 1.356
- Turbine efficiency = 75% (est.)
- Inlet temperature = 1404 °R
- Inlet pressure = 3,480 psia
- Propellants N_2O_4/UDMH at mr = 41.5



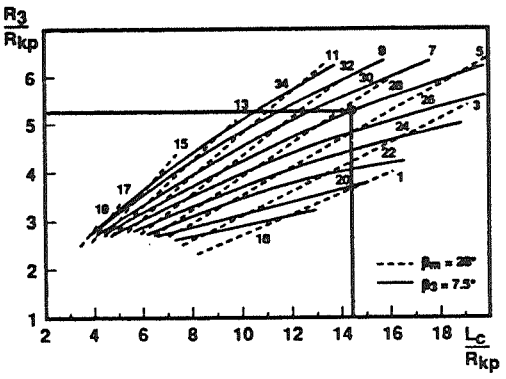
Combustor



Nozzle

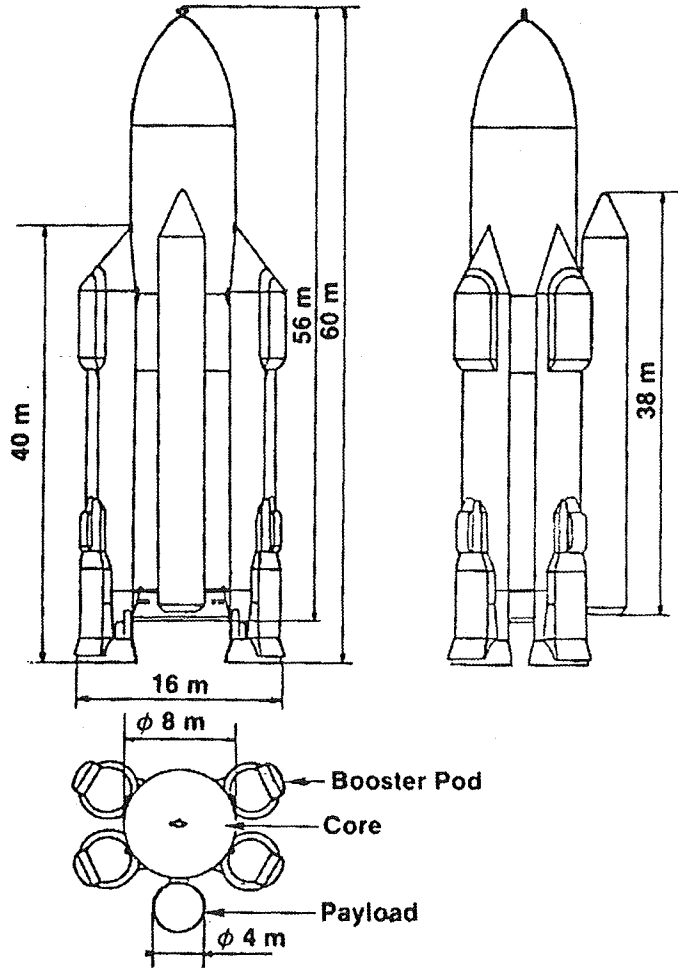


$$\frac{R_3}{R_{kp}} = \sqrt{\varepsilon} = 5.10 \qquad \frac{L_c}{R_{kp}} = \frac{2\,(72.04)}{10.9046} = 13.21$$



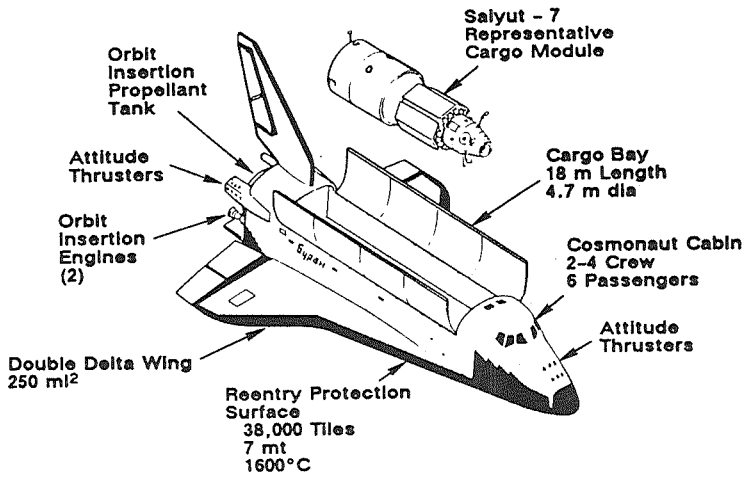
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ENERGIA Views



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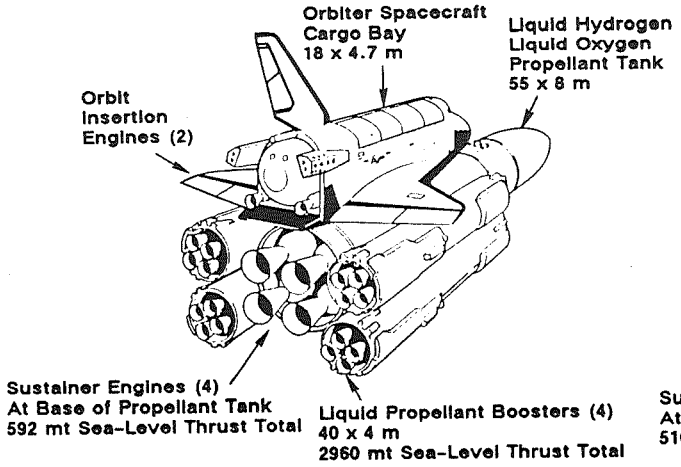
A Comparison of the U.S. and USSR Space Shuttles Reveals Them to be Functional Twins



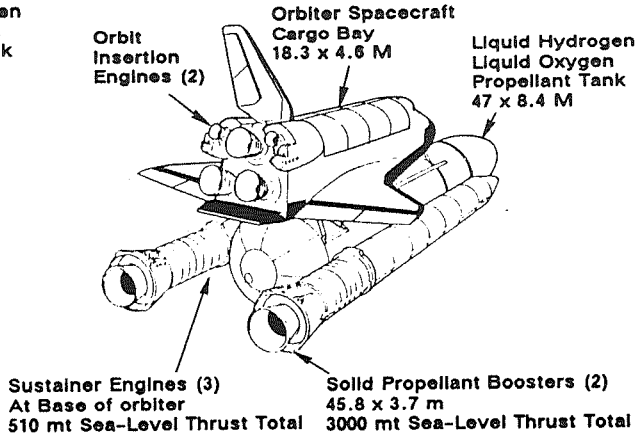
Year of Maiden Flight	Buran 1988	US STS 1981
Orbiter Dimensions		
Length (m)	36.4	37.2
Height (m)	16.5	17.3
Body width (m)	5.6	5.5
Wing span (m)	24.0	23.8
Maximum Mass at Launch		
Orbiter (metric tons)	75	98
Payload (metric tons)	30	24
Total (metric tons)	105	122
Maximum Mass at Landing		
Orbiter (metric tons)	62	86
Payload (metric tons)	20	15
Total (metric tons)	82	100
Total System at Launch		
Length (m)	60	56
Mass (metric tons)	2,400	2,058
Number of main engines	8	5
Thrust (metric tons)	3,552	3,510

(Buran masses are design values;
US STS masses are demonstrated)

Soviet Space Shuttle



U.S. Space Shuttle



Rockwell International
Rocketdyne Division

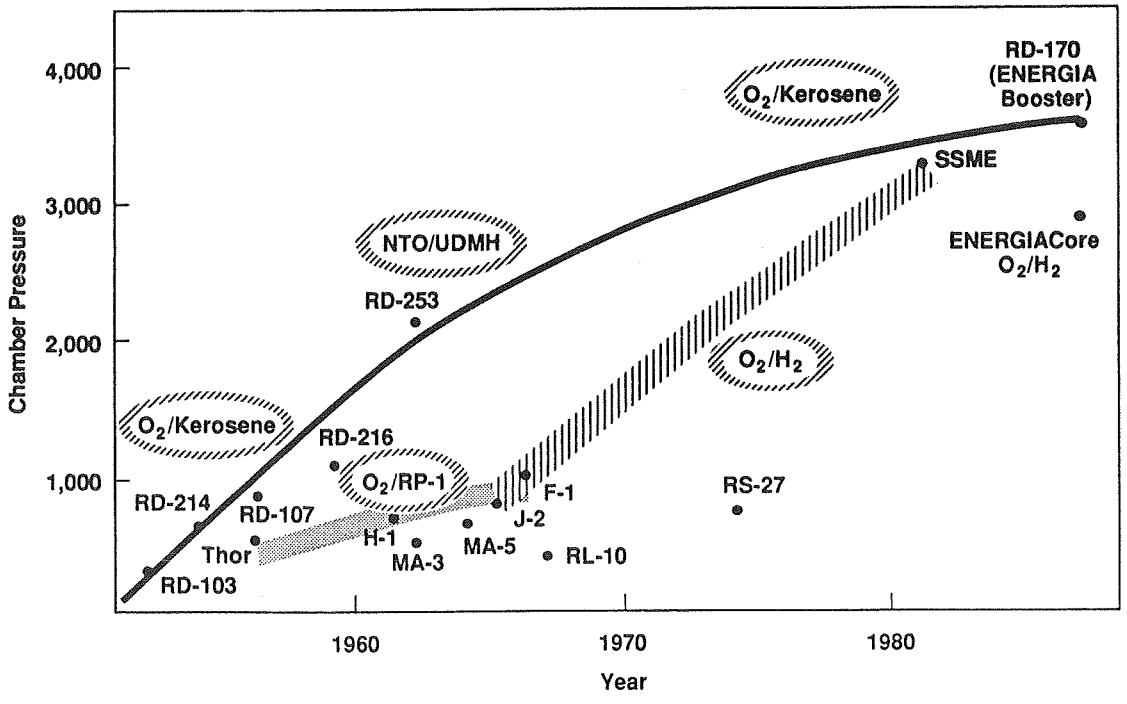
Courtesy: D. R. Woods

Glasnost Provides News of ENERGIA

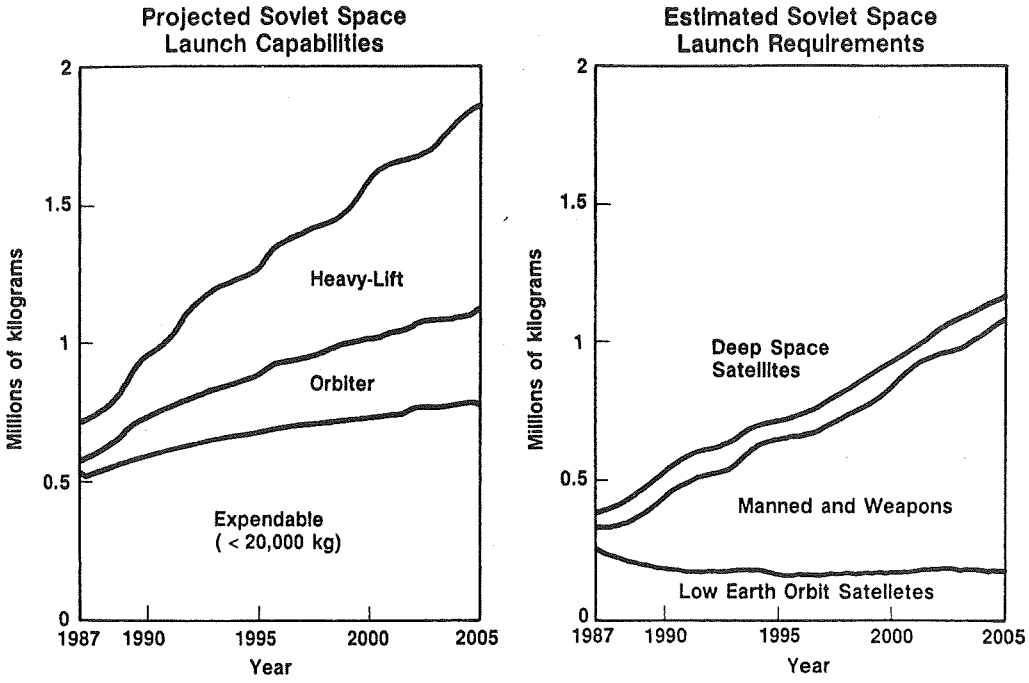
Significant news articles with photographs have been released concerning ENERGIA including substantial quotes from Alexander Duneyev (Glavcosmos head) & B. Gubanov (Chief Designer-ENERGIA). Additionally, a video was shown on Soviet television recently concerning the background of technology & initial launch of ENERGIA & of the promotion video of Cape York usage for the SL-16 (Zenith). Unclassified information was collected from the public domain & used to generate this briefing. The following sources were used:

- Aviation Week & Space Technology
- Soviet Aerospace
- Compendium of Global Launch Vehicles (Rockwell STS Division)
- Data Base - International Space Launchers
- Flight International
- Defense Daily
- Janes Defense Weekly; Janes Intelligence Review
- Teledyne Brown Engineering - Soviet Year in Space (1986, 1987, 1988, 1989)
- Space Magazine
- The Development of Rocket Engineering & Cosmonautics in the USSR, V. Glushko, 1987
- Moscow Pravda articles
- Soviet video - "ENERGIA is Off"
- Soviet video - "Soviet Zenith launcher (Cape York)"
- Paris Air Show display, 1989
- Soviet Military Power 1988, 1989
- Air & Cosmos
- Space markets
- 40th Congress of the IAF (ENERGIA; B. Gubanov)

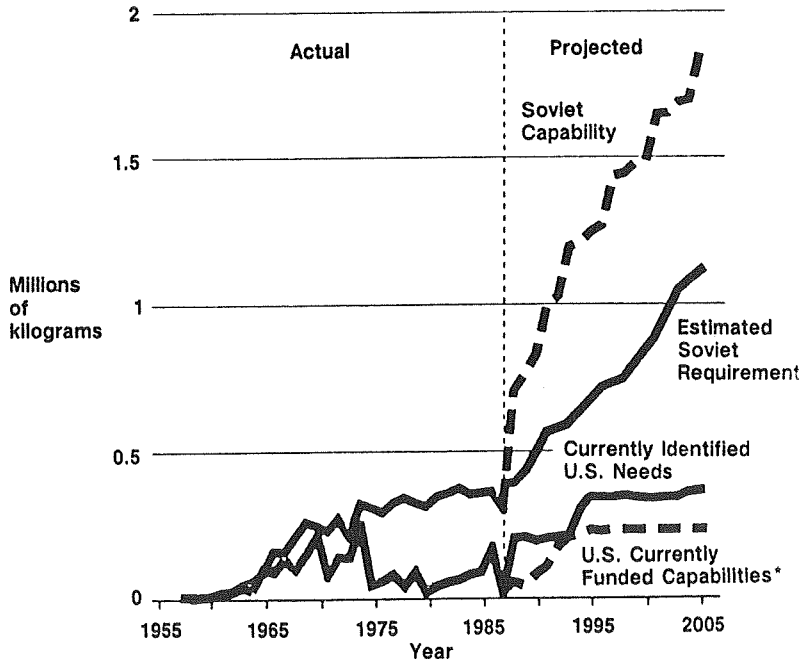
Chamber Pressure Trends in Rocket Engines



Projected Soviet Space Launch Capability Almost Double the Estimated Requirement by 2005



Comparison of United States and Soviet Weight to Orbit Profiles



*Does not include the proposed Advanced Launch System

Rockwell International

ENERGIA — A New Versatile Rocket Space Transportation System

(40th IAF Congress, October 1989, B.I. Gubanov, Glavcosmos, Moscow, USSR)

- Soviet Soyuz opened unmanned & manned space flight era
- Soviet Proton launcher enabled orbiting space stations, vehicles for lunar/planetary study, & means of earth exploration from space
- Soviet VRST system opens new phase of space commercialization
- ENERGIA side-mounted payload allowed independent development of launcher & orbiter
- First stage (booster) has four rocket modules (O_2/kerosene); modules are transported to cosmodrome from manufacturing plant by railway; each module is one rocket engine
- Second stage (core) has four rocket engines (O_2/H_2); core propulsion systems delivered to cosmodrome from manufacturing plant by airplane
- ENERGIA has payload versatility
 - Two booster modules
 - One core module
 } 65 tonne PL
 - Four booster modules
 - One core module
 } 100 tonne PL
 - Eight mod booster modules
 - One core module
 } 200 tonne PL
- ENERGIA features reliability & vitality (life)
 - Key system redundancy exists
 - Turbogenerator power supplies to core (quadrupled)
 - Booster stage batteries (doubled)

ENERGIA — A New Versatile Rocket Space Transportation System

(40th IAF Congress, October 1989, B.I. Gubanov, Glavcosmos, Moscow, USSR)

- ENERGIA development united >1,200 design offices, institutes, plants, assembly organizations, & academies
- 360 test stands/experimental facilities used including
 - 100 aerodynamic models
 - Booster & core engine test stands
 - Module hot fire test in-vehicle capability
 - 10:1 scale-down model of ENERGIA-Buran for launch load study
- 7,000 complex tests & tens of thousands of supporting tests conducted
- More than 100 O_2/H_2 engines manufactured
 - More than 600 engine tests in development (120,000 test-s)
 - Demonstration of 6-7 flight lives
- About 200 O_2/kerosene engines manufactured
 - More than 600 engine tests in development
 - Demonstration of 6-10 flight lives
- Supporting pnemo-hydraulic plumbing included 30,000 tests
- Launcher control system 65,000 tests
- Eight tests of full-scale booster modules & two tests of core modules successful
- Eight full-scale VRST systems produced (five ENERGIA & three ENERGIA-Buran complexes)

ENERGIA – A New Versatile Rocket Space Transportation System

(40th IAF Congress, October 1989,
B.I. Gubanov, Glavcosmos, Moscow, USSR)

- ENERGIA development included new high strength steels, aluminum, & titanium alloys (representing 75% of dry weight)
- The ENERGIA is a heavy lift launcher that will allow delivery of Martian soil to earth & to subsequently perform manned Mars expeditions
- ENERGIA future projections
 - Growth by O_2/H_2 booster stage development
 - For 18 tonne GEO
 - For 32 tonne translunar
 - For 28 tonne Mars
 - A new cargo propulsion system (100 kN thrust, 490 s specific impulse for 5.5 dia stage)
 - Small transport module development (based on existing launcher) for space station placement to 1,000 km
 - An existing cargo propulsion system (O_2/kerosene; possibly Proton Stage IV engine; 85 kN thrust, 350 s specific impulse)
 - Special cargo module development (5.5 in. x 37 in.)
 - New side-mounted universal cargo container
 - General capabilities improvement; economics; reusability aspects
 - Possible new booster propulsion

Soviet SL-17 (ENERGIA) Booster Propulsion

Component Orientation

- The centrally located TPA is a single shaft assembly with high pressure fuel pump on the bottom, high pressure oxygen pump in the middle & turbine on top
- 1989 Paris Air Show display photographs indicate low pressure kerosene inlet ("angle"), low pressure oxygen inlet ("vertical"), preburners ("horizontal"), & four regeneratively cooled (fuel) TCAs.
- Each TCA is two-plane hinged for booster control

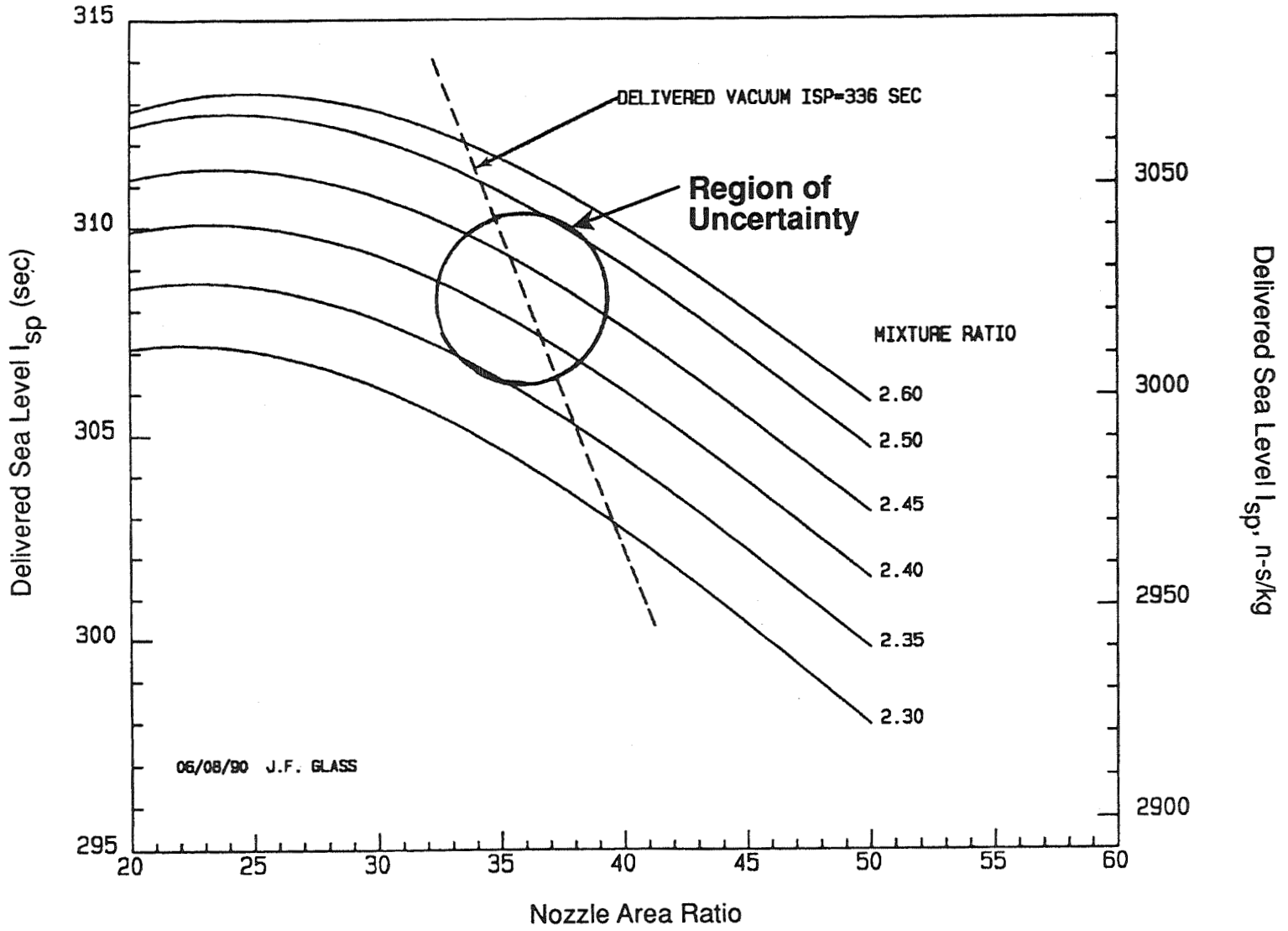
Soviet ENERGIA Booster Propulsion Display
Paris Air Show - 1989
RD-170 Placard

• Le propulseur est installe au premier etage de la fusee "ENERGUYA"	The rocket is installed in the first stage of the "ENERGIA"
• Poussee au sol - 740 ts (metric tons)	Thrust in atmosphere - 1,631,404 lb
• Poussee dans le vide - 806 ts (metric tons)	Thrust in space - 1,776,908 lb
• Impulsion specifique au sol - 308 s	Specific impulse in atmosphere - 308 s
• Impulsion specifique dans le vide - 336 s	Specific impulse in space - 336 s
• Pression dans la chambre de combustion - 250 kgs/cm2	Combustion chamber pressure (3,556 psi)
• Comburant: Oxygene	Oxidizer: Oxygen
• Propergol: Kerozene	Fuel: Kerosene

Soviet SL-17 (ENERGIA) Booster Propulsion

- Glushko Design Bureau has developed world's highest thrust & specific impulse 02/Kerosene engine
 - FSL 1,631,420 lb
 - FV 1,776,928 lb
 - ISSL 308 s
 - ISV 336 s
 - Pc 3,556 psi
 - MR 2.58
- This engine (RD-170) is recognized as a propulsion module & consists of 1 turbopump assembly driven by 2 preburners which feed 4 thrust chamber assemblies
 - Staged combustion power cycle
 - Single shaft TPA centrally located in booster pod
 - System includes low pressure pumps
- The RD-170 engine has flown 29 times
 - Twenty-one as SL-16 (ZENITH) booster since 1985 (21 flights, 1 pod/launcher)
 - Eight as SL-17 (ENERGIA) booster since 1987 (2 flights, 4 pods/launcher

ENERGIA Booster Propulsion Performance Profile
(O_2/Kerosene Specific Impulse As A Function of Area Ratio & Mixture Ratio)



Rockwell International
Rocketdyne Division

Design Comments on RD-170 Photographs

(R. Saxelby, D. Southwick)

- Selected notes as of 25 July 1989
 - Fuel cooled nozzle & MCC
 - Fuel inlet is the one with flat cover. *BW-16, *BW-13
 - It is possible that only the upper half of nozzle is cooled (one pass cooling) *BW-10; others are shown on schematic
 - All of the chamber coolant goes directly to the injector/MCC. None (or very little) leaves the cooling circuit to go to the preburner *BW-12
 - Low pressure fuel boost pump. The pump is liquid driven *BW-16, -17. This is based on the fact that the turbine manifold is small & there isn't a turbine gas outlet
 - Low pressure liquid oxygen boost pump. Manifold *BW-3 shows the manifold feeding a turbine. The mirror view *BW-11, shows one of the feed lines that supply the manifold. A liquid oxygen driver is suggested since there isn't a turbine gas outlet
 - Single turbine/single shaft. No visual evidence of gearbox and/or more than one turbine (turbine could be multiple stages)

*BW = from B. Waldman photographs

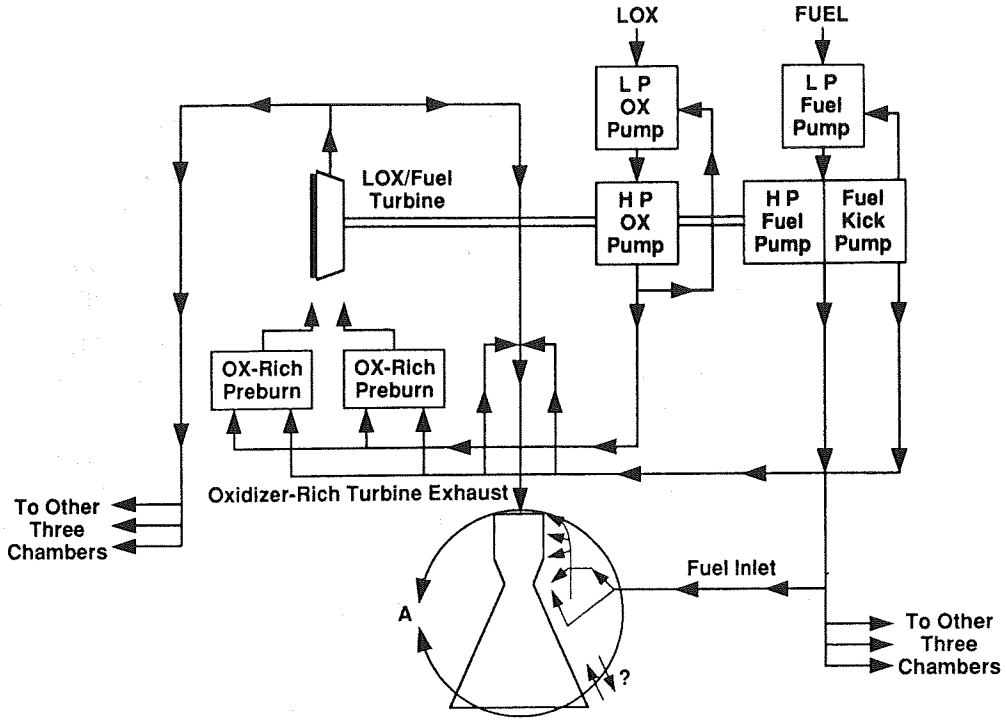
Design Comments on RD-170 Photographs

(R. Saxelby, D. Southwick)

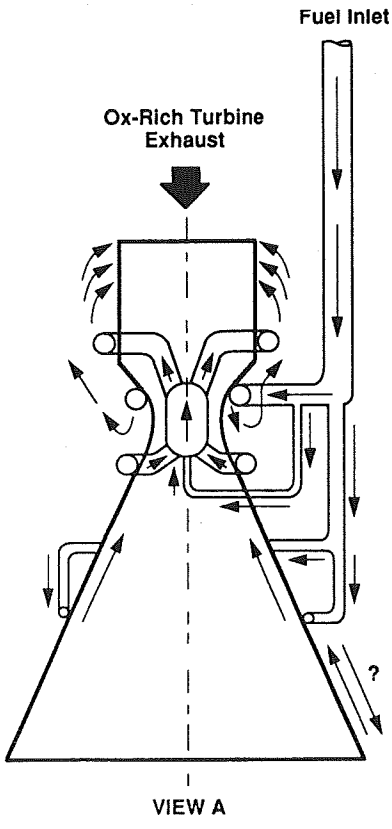
- Selected notes as of 25 July 1989 (continued)
 - High pressure fuel turbopump is located at the bottom of the central unit *BW-16. HP fuel pump inlet is indicated in *BW-12. Main HP outlet goes to top of engines & splits into four pipes - each of which supplies a thrust chamber with coolant. Two smaller tubes leave HP fuel pump *BW-8. There might even be a high pressure fuel kick pump at the very bottom where the two just mentioned smaller tubes leave. Although it cannot be positively shown, it is believed that one line drives LP boost pump & one line goes to the preburners
 - High pressure OX pump is near middle of central unit *BW-2, -3, -19, *BW-15. The outlet of the pump goes to the two preburners *BW-8, -18
 - Turbine is on top of central unit. The turbine exhaust goes directly into the top of each thrust chamber. The inlet to the turbine comes from the straight section of the preburner *BW-15, -18, *BW-6, -4
 - Oxidizer rich preburner. All (or very much) of OX goes into preburner *BW-8
 - LOX pump seal possibly drains back into HP pump inlet. *BW-6, *BW-15

*BW = from B. Waldman photographs

Soviet RD-170 Propulsion System Schematic Diagram
(Based on 1989 Paris Air Show Display Photos)



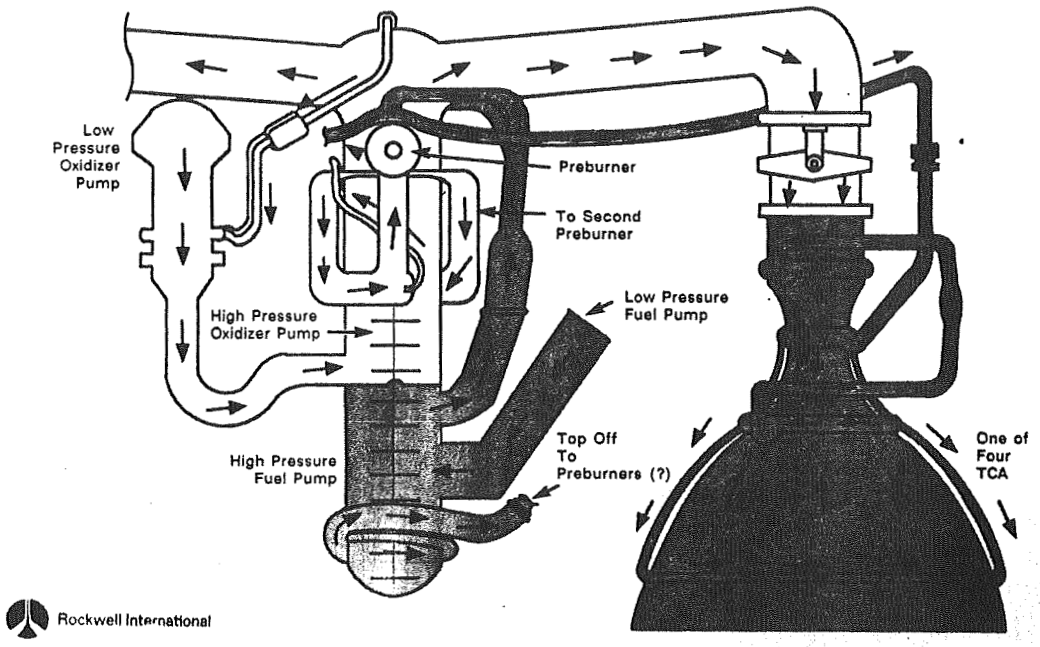
Soviet RD-170 Propulsion System Schematic Diagram
(Based on 1989 Paris Air Show Display Photos)



Rockwell International

Soviet RD-170 Propulsion System Schematic Diagram

(Preliminary Data: WPAFB-FTD 25 July 89)



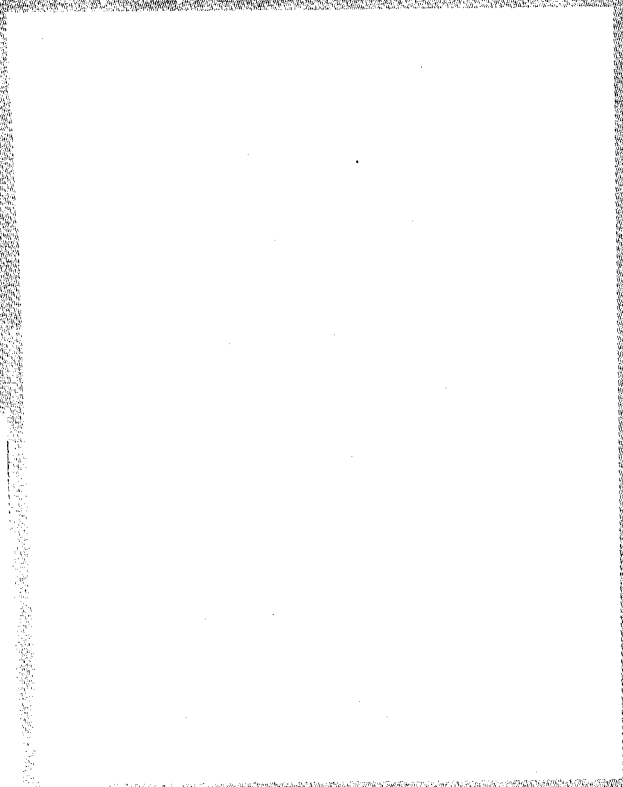
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ENERGIA Booster Propulsion Characteristics Summary

Parameters	Metric Units	English Units
Sea-level thrust (ea TC)	185 t	407,829 lbf
Vacuum thrust (ea TC)	201.5 t	444,223 lbf
Sea-level thrust (ea pod)	740 t	1,631,316 lbf
Vacuum thrust (ea pod)	806 t	1,776,892 lbf
Total booster thrust (sl)	2,960 t	6,525,264 lbf
Total booster thrust (vac)	3,224 t	7,107,568 lbf
Sea-level I_{sp}	3024.9 n-s/kg	308.45 s
Vacuum I_{sp}	3295 N-s/kg	336 s
I_{sp} efficiency	.9438	
Chamber pressure	kg/cm² N/cm²	
Inj	250 2451.7	3556 lb/in.²
Noz	233.68 2382.8	3456 lb/in.²
Area ratio	36	
Throat area	443.81 cm²	68.79 in.²
Delivered c^*	1763.3 m/s	5,785 ft/s
Minimum gimbal angle	None (does not gimbal)	
Engine length	2.524 m	99.372 in.
Engine exit dia (ea)	142.63 cm	56.152 in.
Mixture ratio (o/f)	2.43 (from tanked masses, video, & performance)	
Flow rate, fuel (ea pod)	699.35 kg/s	1541.8 lb/s
Flow rate, oxidizer (ea pod)	1699.41 kg/s	3746.6 lb/s
Horsepower (4 chambers)	186,425 kW	250,000 hp

Booster Pod Propulsion
- One RD-170 engine per booster pod
- Staged-combustion power cycle
- Four thrust chambers fed by one turbopump set



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ENERGIA Booster Engine
Power Balance Analysis by Rocketdyne

- Baseline data
 - O_2/kerosene propellants
 - MR = 2.47 P_c = 3,556 psi
 - F_{sl} = 408 klb/chamber
 - I_{sl} = 308 s (ϵ = 40)
 - From Soviet text translations (Ovsyannikov & Borovskiy)
 - Staged combustion cycle favored for high P_c
 - Oxidizer-rich preburners favored (if fuel is not H_2)
 - Turbine pressure ratio = 1.3 - 1.8 for staged combustion
 - Maximum turbine inlet temperatures = 2160°R for fuel rich; 1440°R for oxidizer rich
- RD-170 simulation
 - Advanced heat transfer
 - Fuel-rich & oxidizer-rich preburners can meet 3,600 psia P_c
 - Within temperature limits
 - Requires kick pump stages and/or boost pumps
 - Mixed preburners can exceed 3,600 psia P_c
 - Within temperature limits
 - With or without boost pumps
 - Turbine pressure ratio ≈2.0
 - Pump ΔPs ≈12,000 psi → 9000 psi
 - $I_{sp_{vac}}$ = 339 with ηc^* = 0.96
 - It is noted fuel-rich preburners would tend to plug main injector (oxidizer-rich preburners would not)

RD-170 Turbopump Configurations Evaluated to Yield 3,600 psia P_c

Case	Pre-burner	Shafts	Main Pump Stages		Kick Pump Stages		Boost Pumps		Required Turbine Inlet Temperature (°R)
			Fuel	Oxidizer	Fuel	Oxidizer	Fuel	Oxidizer	
1	FR	1	2	2	0	0	0	0	2390
2	FR	1	2	1	1	1	0	0	2130 ✓
3	OR	1	3	2	0	0	0	0	1760
4	OR	1	1	2	1	0	0	0	1680
5	OR	1	1	2	2	0	0	0	1540
6	OR	1	1	2	2	0	1	1	1440 ✓
7	Mixed	2	2	2	0	0	0	0	1840/1440 (F/O)
8	Mixed	2	2	2	0	0	1	1	1750/1250 (F/O)

✓ Acceptable

Preliminary Engine Balance Data for RD-170 Booster Engine

(Selected Turbomachinery Parameters)

		OXIDIZER	FUEL	HYDROGEN
TYPE	(NONE)	PRESSURE	REACTION	PRESSURE
# OF STAGES	(NONE)	2.00	2.00	.00
HORSEPOWER	(HP)	.	294,400	.00
FLOWRATE	(LB/SEC)	.000	3817	.00000
EFFICIENCY	(NONE)	.00000	.83837	.00000
PRESSURE RATIO	(NONE)	2.227	2.227	.000
ADMISSION	(FRACTION)	1.000	1.000	.000
VELOCITY RATIO	(NONE)	.000	.501	.000
PITCH DIAMETER	(IN)	.000	19.54	.000
1ST STG BLADE HEIGHT	(IN)	.000	2.672	.000
2ND STG BLADE HEIGHT	(IN)	.000	4.316	.000
PITCHLINE VELOCITY	(FT/SEC)	1061.85	904.45	.00
INLET HUB/TIP RATIO	(NONE)	.000	.759	.000
EXIT HUB/TIP RATIO	(NONE)	.000	.638	.000
TIP SPEED	(FT/SEC)	.00	1104.17	.00
BEARING DN*E-6	(MM*RPM)	.000	1.910	.000
ANNULUS AREA*N**2*E-10	((IN*RPM)**2)	.000	2.976	.000
INLET PRESSURE	(PSIA)	9626.18	9626.18	.00
OUTLET PRESSURE	(PSIA)	4322.17	4322.17	171.69
INLET TEMPERATURE	(DEG R)	.00	1440.00	1440.00
OUTLET TEMPERATURE	(DEG R)	.00	1218.59	.00
1ST BLADE TEMPERATURE	(DEG R)	.00	1380.22	.00
2ND BLADE TEMPERATURE	(DEG R)	.00	1269.51	.00
MOMENT OF INERTIA	(FT-LB-SEC**2)	.075		.000
TIME CONSTANT	(SEC)	.000		.000

Preliminary Engine Balance Data for RD-170 Booster Engine

(Selected Turbomachinery Parameters)

PUMP DESCRIPTION	(UNITS)	MAIN PUMP OXIDIZER	FUEL	KICK PUMP OXIDIZER	FUEL
# OF STAGES	(NONE)	2.00	1.00	.00	2.00
HORSEPOWER	(HP)	205,500	49,770		39,140
ROTATING SPEED	(RPM)	10,600	10,600		10,600
EFFICIENCY	(NONE)	.80724	.76478	.00000	.73933
INLET PRESSURE	(PSIA)	65.00	45.00	.00	4824.52
OUTLET PRESSURE	(PSIA)	12091.69	4824.52	.00	12091.69
FLOWRATE	(LB/SEC)	3752	1519		760
	(GPM)	23,650	13,650		6,827
INDUCER					
TIP DIAMETER	(IN)	13.36	11.12		
TIP SPEED	(FT/SEC)	617.90	514.55		.00
INLET FLOW VELOCITY	(FT/SEC)	61.75	51.42		40.81
FLOW COEFFICIENT	(NONE)	.100	.100	.000	.100
IMPELLER					
TIP DIAMETER	(IN)	19.34	19.74		17.00
TIP SPEED	(FT/SEC)	895.45	914.01	.00	786.75
TIP WIDTH	(IN)	1.46	.972		.704
HEAD COEFFICIENT	(NONE)	.488	.531	.000	.545
BLADE ANGLE	(DEG)	25.000	25.000	25.000	25.000
HEAD RISE (OVERALL)	(FT)	24313.39	13781.57	.00	20954.58
STAGE SPECIFIC SPEED	(RPM*GPM**.5/FT**.75)	1407.70	973.66	.00	845.62
HEAD RISE (ISENTROPIC)	(FT)	24313.39	13781.57	.00	20954.58
EFFICIENCY (ISENTROPIC)	(NONE)	.80724	.76478	.00000	.73933
HUB/TIP RATIO	(NONE)			.000	.301

Rocketdyne Power Balance Analysis Conclusions for RD-170 Engine

- ENERGIA booster engine must be staged combustion to obtain 308 sec I_{sp} at 3,556 psi P_c
- Oxidizer-rich preburner avoids injector plugging
- Fuel kick pump helps maximize energy utilization
- Boost pumps help obtain high pump η at low tank pressures
- Power balance analysis doesn't answer 1 vs 4 tip set/pod question; but Paris Air Show 1989 display does!

Heat Transfer Considerations - RD-170 Booster Engine

- **Severe engine operating conditions**
- **Regenerative cooled construction**
- **Candidate coolants**
 - LO_2
 - LH_2 (from core engine)
 - Kerosene
- **Methods for reducing heat flux & pressure drop losses**
 - Ceramic coatings
 - Fuel-rich outer zone (carbon layer)
 - Silicon oil additive

RD-170 Propulsion System Cooling Feasibility Verified

- P_c = 3,556 psia
- O_2/kerosene (MR = 2.47)
- Nozzle 40:1

Heat transfer analysis

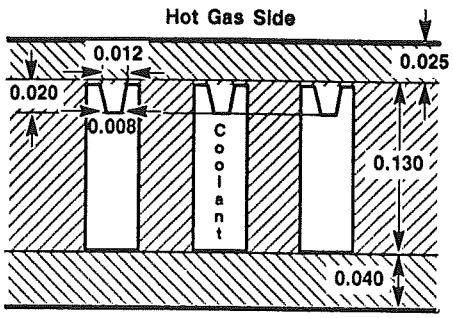
- Advanced heat transfer analysis applied
- Data base from NAS8-3657
- Modeled small throat radius ratio SSME geometry
- NARloy-Z chamber with milled, finned channels & electroformed closeout
- Peak q/Δ at 70 Btu/in²-s (ϵ_c = 2.66, ϵ_e = 5)
- 430 channel design (0.035 in. width x 0.130 in. depth)

Kerosene as regenerative coolant

- Coking limit 1200°R (high fuel velocity < 500 ft/s)
- Hot gas wall limit 1600°R (Cu-alloy)
- Bulk coolant limit 900°R
- Fuel pump discharge pressure < 12,000 psi

Coatings, fuel-rich bias injection, & silicon oil additive also considered

Channel Geometry



Closeout Section

RD-170 Cooling Feasibility Throat Parameters Comparison
(Finned Channel Construction)

- Kerosene as coolant

Case	Q/A* (B/In.²-s)	%hg* (B/In.²-s-F)	Twg* (°F)	Twc* (°F)	$\dot{w}_c$ (#/s) 65.4%	ΔP (#/In.²)	ΔTc (°F)	Vc* (#/s)
1	75.6	100	1205	798	250	2,320	219	409
2	69.1	90	1126	755	250	2,319	202	408
3	62.4	80	1035	711	250	2,318	184	408
(4)	55.5	70	960	664	250	2,320	166	407

- Kerosene as coolant plus coating

Case	Coating (in.)	%hg*	hg*
0	0	100	0.151
1	0.001	87.9	0.01327
2	0.002	78.9	0.01191
(3)	0.003	71.5	0.0108
4	0.004	65.4	0.00988

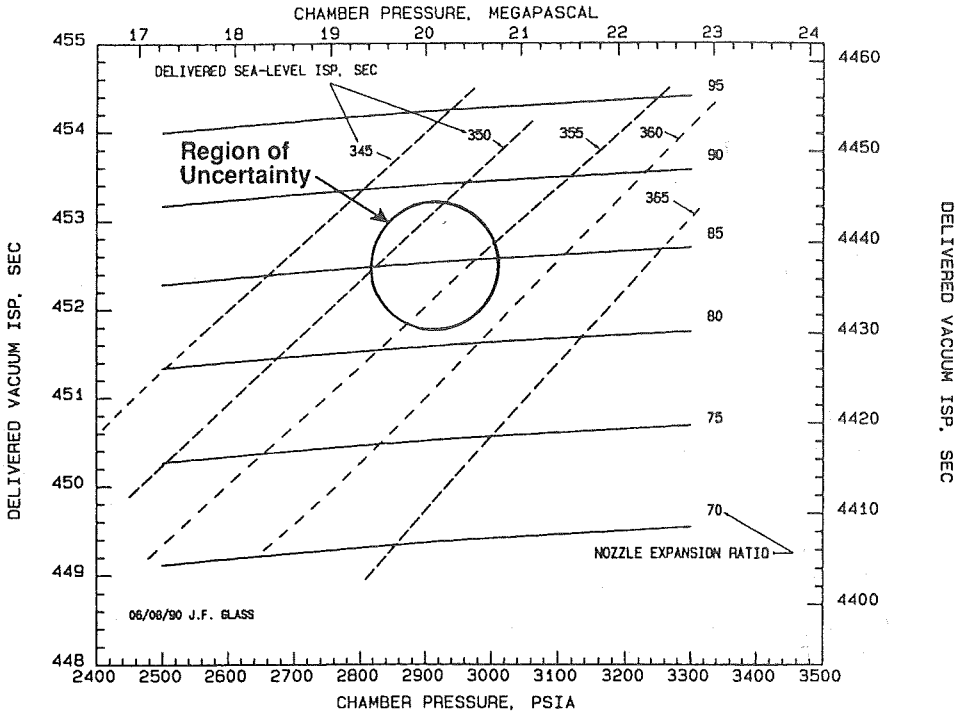
- Kerosene as coolant plus film cooling

Case	$\%\dot{w}_f$	$\%\dot{w}_g$	%hg*
1	0	0	100
2	5	1.49	85
(3)	10	2.98	70
4	15	4.48	60
5	20	5.97	50
6	25	7.46	40

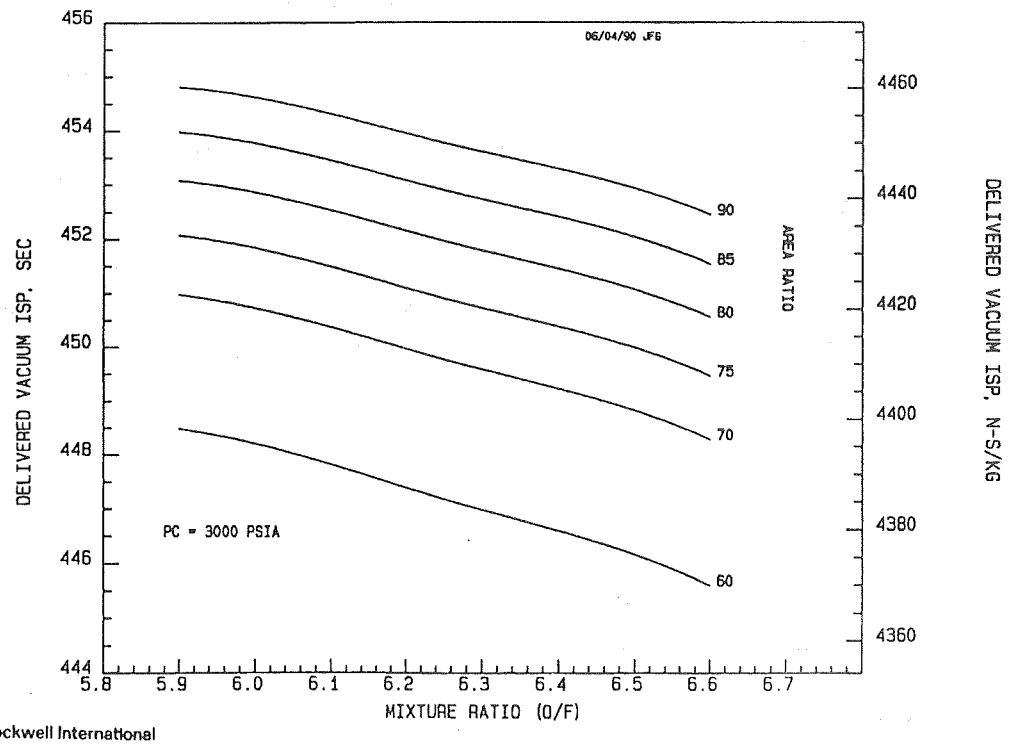
*K = 5 Btu/hrft°F
Nicraly-Zirconia Mix

Rockwell International

ENERGIA Core Propulsion Performance Profile
(O_2/H_2 Specific Impulse As A Function of Chamber Pressure & Area Ratio)



ENERGIA Core Propulsion Performance
I_{svac} As A Function of Mixture Ratio For Several Area Ratio Values



Rockwell International

Soviet ENERGIA O2/H2 Core Engine Design Comments
(Based on WPAFB-FTD Photographs)

- Most likely power cycle is staged combustion (performance basis)
- Single preburner or top of main combustor drives both pumps
- Nozzle has full external structural jacket with hatbands on lower 2/3 only. The half-round, closer spaced hatbands are insulated; less costly, but less efficient than the square hatbands. (Spot welds indicate thick jacket)
- Nozzle cooling via single up-pass (two inlets at nozzle exit) followed by dump cooling at forward end with low-pressure coolant (supports high operating mixture ratio & double wall design). Nozzle inside surface is smooth (nontubular construction).
- Heat shield support structure with drain/feedline penetrations is continuous ring with webbed load ribs

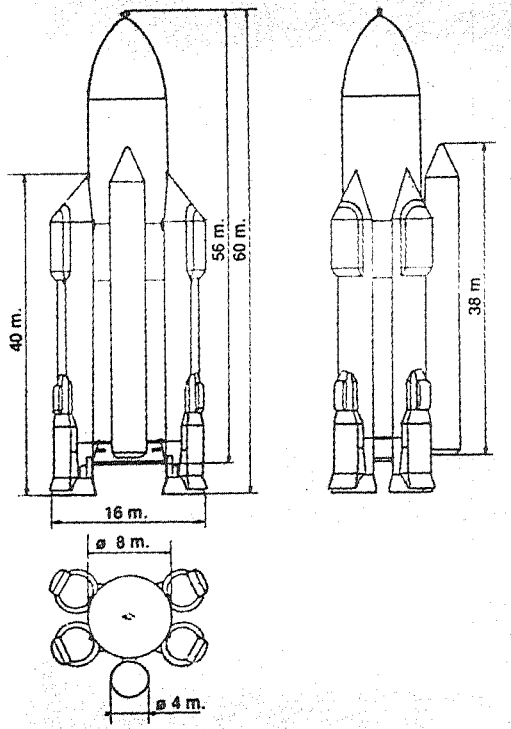
Soviet ENERGIA O2/H2 Core Engine Design Comments
(Based on WPAFB-FTD Photographs)

- Drain lines run parallel for nozzle coolant feedlines (angled to accommodate thermal movement). Two lines are insulated (possibly hydraulic 0:1). Numerous lines indicates multiple turbopumps involved in power pack
- Considerable use of insulation on components/ducting. Brown (polyurethane) & white (sealant) are wrapped/ sprayed on
- Small white canister shapes may be SPGG units or electromechanical actuator or motors for preburner or coolant valves
- Shaped cylinder could be for POGO suppressor, or pressure bottle for inert gas for turbopump

ENERGIA Vehicle Characteristics Summary

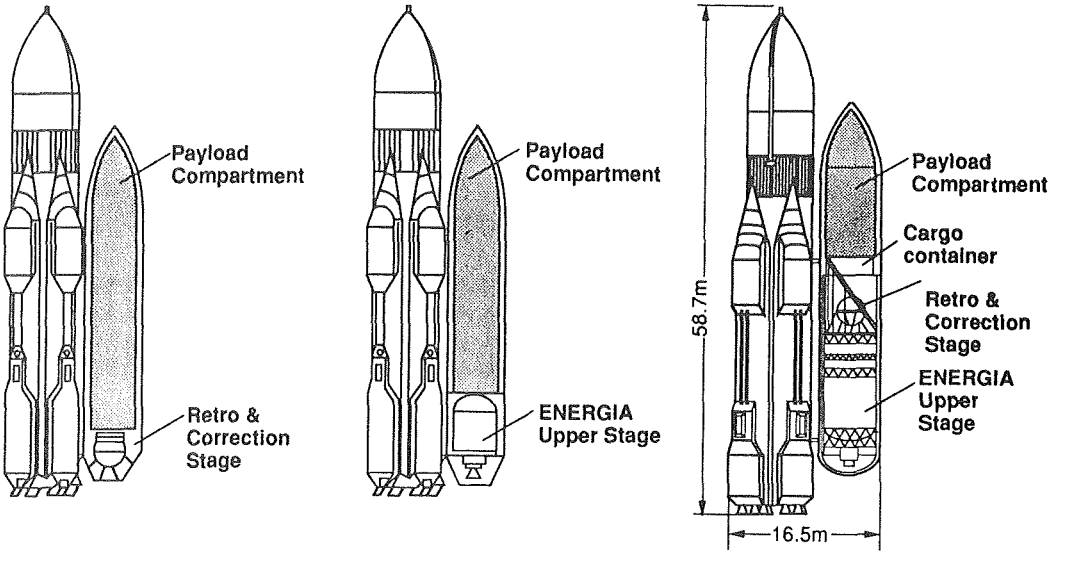
Vehicle Characteristics

Parameters	Metric Units	English Units
Min. liftoff accel	1.48 g	
Max. liftoff accel	1.77 g	
Max. liftoff mass	2,400 t	5,291,040 lb
Max. liftoff thrust	3,552 t	7,830,739 lbf
Duration, core section	455.0 s	7.583 min
Duration, booster section	137.4 s	2.289 min
Total vehicle ΔV	8814.3 m/s	28,918 ft/s
ΔV during booster burn	3514 m/s	11,529 ft/s
ΔV during core-only	5300.4 m/s	17,389 ft/s
Average booster I_{sp}	3185.2 N-s/kg	324.8 s
Average core I_{sp}	3998.2 N-s/kg	407.7 s
Core fuel load (MR = 6)	114.29 t	251,954 lb
Core oxidizer load (MR = 6)	685.71 t	1,511,724 lb
Booster fuel load (MR = 2.47)	379.83 t	837,367 lb
Booster oxidizer load (MR = 2.47)	938.17 t	2,068,296 lb
Total fuel load	494.12 t	1,089,337 lb
Total oxidizer load	1623.88 t	3,580,006 lb
Vehicle length	60 m	196.85 ft
Vehicle diameter	16 m	52.49 ft
Core diameter	8 m	26.25 ft
Booster diameter	4 m	13.12 ft
Booster length	40 m	131.23 ft
Payload diameter	4 m	13.12 ft
Payload length	38 m	124.67 ft
Payload capability (min)		
LEO	100 t	220,000 lb
GEO	18 t	39,683 lb
Lunar trajectory	32 t	70,547 lb
Martian trajectory	28 t	61,729 lb



Rockwell International
Rocketdyne Division

ENERGIA Mission Growth Capability Via Cargo Container Propulsion



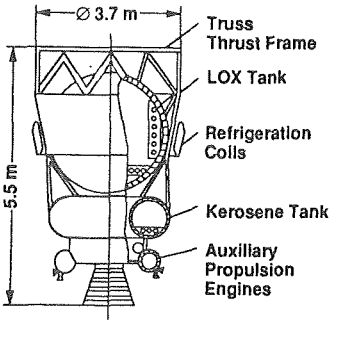
• Glavkosmos diagrams of ENERGIA with its planned cargo container, which will be 42m long & 6.7m in dia. The configuration at left, with the RCS stage alone, is for low Earth orbit missions, with a payload up to 35m long. With the EUS alone, a 23.5m payload can be sent to GEO, lunar libration points or lunar orbit. With both upper-stage motors (right) a 19.5m payload can be accomodated, primarily for planetary orbit or lander missions. Maximum cargo weight (including upper stages) is 93t. Maximum payload dia in all cases is 5.5m. Gross lift-off mass is given as 2,400 tonnes. (All drawings: courtesy of Glavkosmos/Space Commerce Corp joint venture)

Ref: Space Markets 1/1990

ENERGIA Cargo Module Propulsion

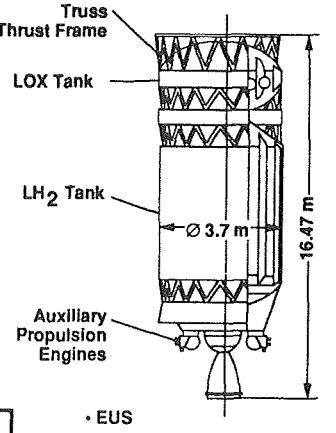
(Space Markets, 1/90; Credit Glavkosmos/Space Commerce Corp.)

ENERGIA's Next Stage, P.S. Clark



• RCS

- Retro & correction stage O_2 /kerosene propellants
- Propulsion could be Proton Block D/DM 4th stage
- For PL to LEO & AS interorbit tug to 1,500 km



• EUS

- ENERGIA upper stage O_2 /H_2 propellants
- For PL to high orbit; geostat & transplanetary

	RCS	EUS
Dry mass, tonnes	Approximately 2	Approximately 7[1]
Maximum useful propellant mass (tonnes)	15	70
Main engine maximum vacuum thrust (kN)	85	100[2]
Main engine specific impulse (s)	Approximately 352[3]	Approximately 490[4]
Maximum number of engine starts	7	10
Maximum engine operating life in space	2 yr	4 days

• Notes: [1] Estimate (see text); [2] Engine can be throttled back to 75kN
[3] Figure derived from apparently similar Proton DM stage;
[4] Figure estimated from performance requirements

Ref: *Space Markets, January 1990*

Rockwell International

EUROPEAN AND OTHER TECHNOLOGY

97

540730
461

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OVERVIEW

of

EUROPEAN AND OTHER NON-US/USSR/JAPAN LAUNCH VEHICLE AND PROPULSION TECHNOLOGY PROGRAMS

Presented

by

Dr. Eric E. Rice
President and Chief Executive Officer



Orbital Technologies Corporation
Madison, Wisconsin

Space Transportation Propulsion Technology Symposium
The Pennsylvania State University, University Park, PA
25-29 June 1990

Space Transportation & Propulsion Technologies Reviewed

* Europe (ESA)	Ariane Family & Hermes Space Plane
* Germany	Sanger Aerospace Plane
* United Kingdom	Hotol Aerospace Plane
* France	Star H Aerospace Plane
* China	Long March Family
* India	SLV, ASLV, PSLV & GSLV
* Italy	Advanced Small Launch Vehicle (ASLV)
* Israel	Shavit
* Norway	LittLEO
* Iraq	ABID
* South Korea	New Initiative
* Brazil	Cancelled Program

Summary of Europe's Advanced Propulsion Technology Activities

* Majority of Propulsion Technology Development Work Is Directly Related to the ESA's Ariane 5 Program and Heavily Involves SEP in All Areas:
 - Vulcain H/O Engine Is a Major Development Led by SEP; 1st Ignition Sequence in 7/90; 1st Full Power Firing in 12/90
 - Performance Improvements Underway, Including Thrust and Combustion Pressure Increases
 - Solid Propulsion Being Expanded, as SEP & BPD Have Formed Europulsion Company Headquartered in Paris for the Main Purpose of Building Large Solids for Ariane 5; Trying to Reduce Costs and Improve Reliability--Like ALS Objectives, But Not as Ambitious.
 - Man Rating of Ariane 5 for Hermes Flights Will be Accomplished in Parallel With Flights of Unmanned Ariane 5 Flights
* Hermes
 - Composite Applications in Small Storable Rocket Engines (Hermes ACS); Have Accomplished 10,000 s of Firing in 200 N Class Engine
* Advanced Work on Magnetic Bearings for Turbomachinery
* Electric Propulsion, Using Cs and Xe Propellants Being Done By SEP in France, MBB ERNO in West Germany, and by Culham Lab in UK

Summary of Europe's Advanced Propulsion Technology Activities (Cont.)

* Successfully Test Fired H/O Composite (Carbon/Silicon Carbide) Nozzle Exit Cone on 3rd Stage of Ariane (HM7)

* Turbine Blades Made of Composites to Allow Increase in Gas Temperature and Improvement in Efficiency

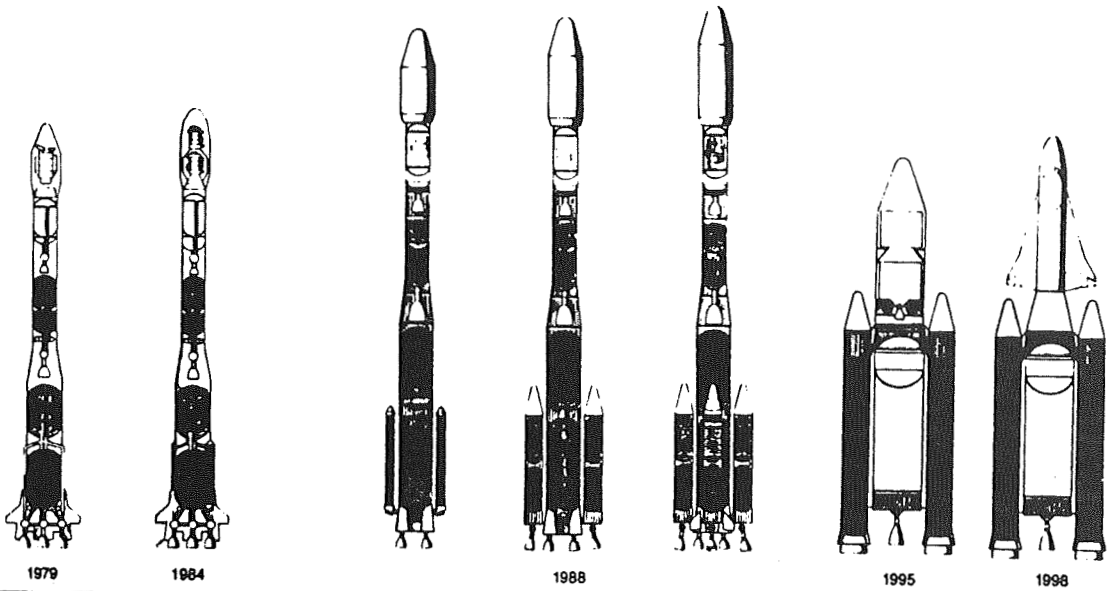
* Combined Cycle (Turboramjet/Rocket) Engine Analysis Work Being Done by Hyperspace, a New Joint Effort of SEP and SCNECMA

* SEP Looking At Future Launchers By Conducting Studies to Determine Advantages of Expendable vs Reusable; Manned vs Unmanned; and Solids vs Liquids

* ESA Advanced Program Studies Looking Beyond Ariane 5: What Payloads Will be Needed in the Future? What Cost Reductions Are Possible? What is Needed For Manned Flight?

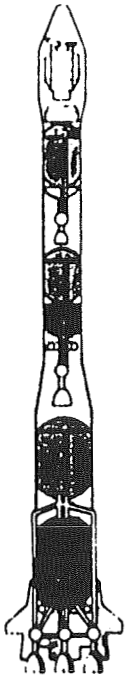


European (ESA) Ariane Family



1979	1984		1988		1995	1998
ariane 1	**ariane 3**	**ariane 4** (42P)	**ariane 4** (44LP)	**ariane 4** (44L)	**ariane 5** (L-5)	**ariane 5** (Hermes)
1835 kg*	2600 kg*	2600 kg*	3700 kg*	4200 kg*	6.8 tonnes*	21 tonnes in low-Earth orbit

ARIANE 1
SUMMARY



STATUS: INACTIVE

1ST LAUNCH: 1979

LAST LAUNCH: 1986

DRY MASS: 21 MT

LIFT-OFF MASS: 210 MT

PAYLOAD MASS INTO GTO: 1760 kg

11 FLIGHTS

2 FAILURES

ARIANE 1

VEHICLE STAGE CHARACTERISTICS

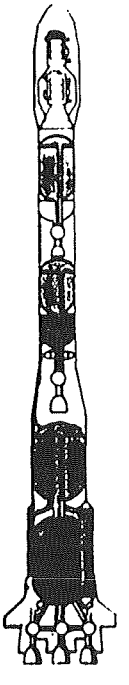
STAGE NO./ MANUFACTURER	1 AEROSPATIALE	2 MBB/ERNO	3 AEROSPATIALE	EQUIPMENT BAY/FAIRING MATRA/ CONTRAVES
STAGE DESIGNATION	L140	L33	H8	
LIFTOFF MASS (kg)	161,000	37,500	9700	323/842
PROPELLANT MASS (kg)	147,600	34,100	8230	
TOTAL THRUST (kN)	2480	726	61	

ENGINE DATA

ENGINE DESIGNATION	Viking V (4)	Viking IV (1)	HM-7 (1)	
ENGINE MANUFACTURER	SEP	SEP	SEP	
THRUST PER ENGINE (kN)	620	726	61	
APPROXIMATE Isp (s)	280	290	425	
PROPELLANTS	UDMH N2O4	UDMH N2O4	LH2 LO2	

ARIANE 2

SUMMARY



ARIANE 2 VEHICLE

LIFT-OFF MASS: 217 MT

DRY MASS: 20.5 MT

PAYLOAD MASS INTO GTO: 2175 kg

ARIANE 2/3 LAUNCH RECORD

STATUS: INACTIVE

1ST LAUNCH: 1984

LAST LAUNCH: 1989

17 FLIGHTS

2 FAILURES

ARIANE 2

VEHICLE STAGE CHARACTERISTICS

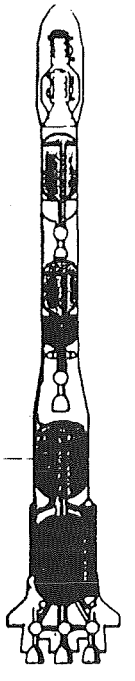
STAGE NO./ MANUFACTURER		1 AEROSPATIALE	2 MBB/ERNO	3 AEROSPATIALE
STAGE DESIGNATION		L140	L33	H8
LIFTOFF MASS (kg)		162,000	39,000	12,300
PROPELLANT MASS (kg)		147,600	35,100	10,700
TOTAL THRUST (kN)		2690	785	63

ENGINE DATA

ENGINE DESIGNATION		Viking V (4)	Viking IV (1)	HM-7B (1)
ENGINE MANUFACTURER		SEP	SEP	SEP
THRUST PER ENGINE (kN)		672	785	63
APPROXIMATE Isp (s)		280	290	435
PROPELLANTS		UH25/H2O N2O4	UH25/N2O4 N2O4	LH2 LO2

ARIANE 3

SUMMARY



ARIANE 3 VEHICLE

LIFT-OFF MASS: 236.8 MT

DRY MASS: 25.5 MT

PAYLOAD MASS INTO GTO: 2580 kg

ARIANE 2/3 LAUNCH RECORD

STATUS: INACTIVE

1ST LAUNCH: 1984

LAST LAUNCH: 1989

17 FLIGHTS

2 FAILURES

ARIANE 3

VEHICLE STAGE CHARACTERISTICS

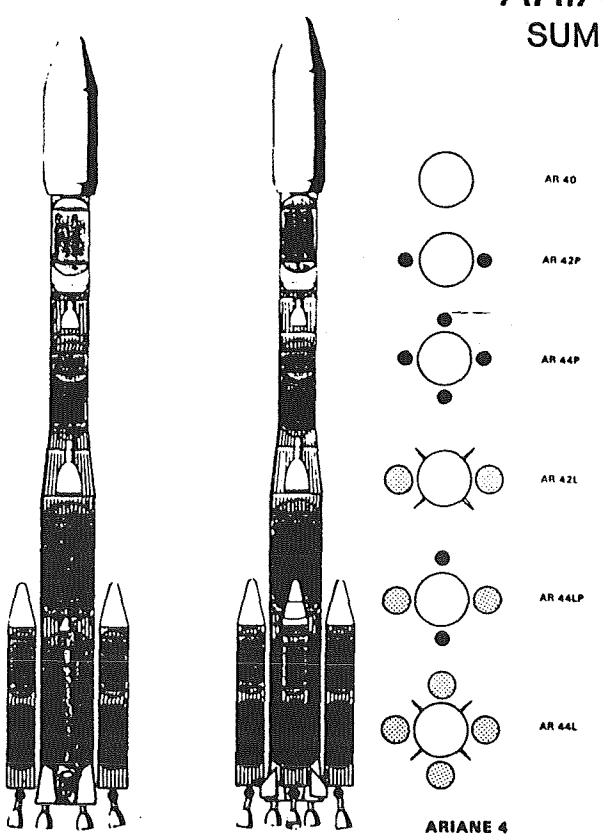
STAGE NO./ MANUFACTURER	0 AEROSPATIALE	1 AEROSPATIALE	2 MBB/ERNO	3 AEROSPATIALE
STAGE DESIGNATION	BOOSTERS	L140	L33	H8
LIFTOFF MASS (kg)	9750 x 2	162,000	39,000	12,300
PROPELLANT MASS (kg)	7350 x 2	147,600	35,100	10,700
TOTAL THRUST (kN)	600 x 2	2690	785	63

ENGINE DATA

ENGINE DESIGNATION	SOLID BOOSTERS	Viking V (4)	Viking IV (1)	HM-7B (1)
ENGINE MANUFACTURER	SNIA/BPD	SEP	SEP	SEP
THRUST PER ENGINE (kN)	600	672	785	63
APPROXIMATE Isp (s)	230	280	290	435
PROPELLANTS	SOLID CTPB 1613	UH25/H2O N2O4	UH25/H2O N2O4	LH2 LO2

ARIANE 4

SUMMARY



ARIANE 4

STATUS: ACTIVE

1ST LAUNCH: 1988

8 FLIGHTS

1 FAILURE: H2O FLOW TERMINATION

SIX STRAP-ON CONFIGURATIONS

CONFIGURATION	GTO PAYLOAD (kg)
40	1900
42P	2600
44P	3000
42L	3200
44LP	3700
44L	4200

ARIANE 4

VEHICLE STAGE CHARACTERISTICS

STAGE NO./ MANUFACTURER		1 * AEROSPATIALE	2 MBB/ERNO	3 AEROSPATIALE
STAGE DESIGNATION		L220	L33	H10
LIFTOFF MASS (kg)		243,000	37,600	11,900
PROPELLANT MASS (kg)		226,000	34,000	10,700
TOTAL THRUST (kN)		2690	785	63

*ADDITIONAL THRUST MAY BE PROVIDED BY STRAP-ONS

ENGINE DATA

ENGINE DESIGNATION		Viking V (4)	Viking IV (1)	HM-7B (1)
ENGINE MANUFACTURER		SEP	SEP	SEP
THRUST PER ENGINE (kN)		672	785	63
APPROXIMATE Isp (s)		280	290	435
PROPELLANTS		UH25/H2O N2O4	UH25/H2O N204	LH2 LO2

ARIANE 4 CONFIGURATIONS

STAGE 1 BOOSTER INFORMATION

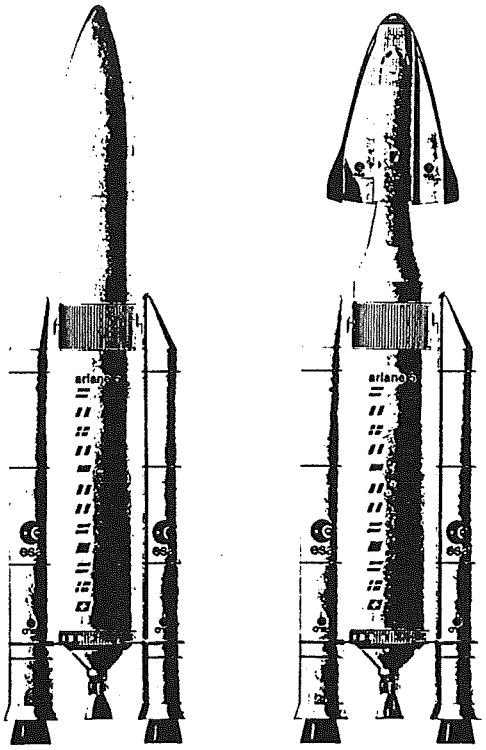
	SOLID	LIQUID
BOOSTER DESIGNATION	PAP	PAL
LIFTOFF MASS (kg)	12,700	43,500
PROPELLANT MASS (kg)	9500	39,000
TOTAL THRUST (kN)	625	666
MANUFACTURER	SNIA-BPD	MBB-ERNO/SEP
APPROXIMATE Isp (s)	230	235
PROPELLANTS	CTPB 1613	UH25-H2O/N2O4

STAGE1/BOOSTER CONFIGURATIONS

DESIGNATION	40	42P	44P	42L	44LP	44L
BOOSTERS	NONE	2 PAP'S	4 PAP'S	2 PAL'S	2 PAP'S 2 PAL'S	4 PAL'S
LIFT-OFF THRUST (kN)	2650	3950	5200	4020	5270	5350
PAYLOAD TO GTO (kg)	1900	2600	3000	3200	3700	4200

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ARIANE 5
SUMMARY



STATUS: IN DEVELOPMENT

1ST LAUNCH: 1998

ARIANE 5

VEHICLE STAGE CHARACTERISTICS

STAGE NO./ MANUFACTURER	0 EUROPROP.	1 AEROSPATIALE	2 MBB/ERNO	
STAGE DESIGNATION	P 230	H 150	L 7	
LIFTOFF MASS (kg)	270,000 x 2	169,000	8130	
PROPELLANT MASS (kg)	230,000 x 2	156,000	7200	
TOTAL THRUST (kN)	6000 x 2	1025	28	

ENGINE DATA

ENGINE DESIGNATION	P 230	VULCAIN (1)		
ENGINE MANUFACTURER	EUROPROP.	SEP	MBB/ERNO	
THRUST PER ENGINE (kN)	6000	1025	28	
APPROXIMATE Isp (s)				
PROPELLANTS	HTPB/AP/AL	LH2 LOX	MMH N2O4	

ARIANE 5 CONFIGURATIONS

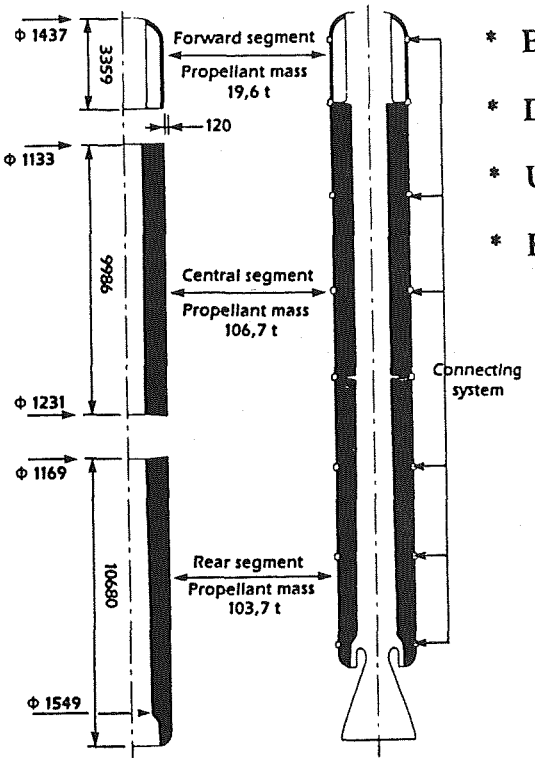
	DOUBLE LAUNCH	SINGLE LAUNCH	HERMES LAUNCH
LIFT-OFF MASS (kg)	721,000	721,000	735,000
LIFT-OFF THRUST (kN)	13,043	13,043	13,043
HEIGHT (m)	49.6	50.4	49.6
PAYLOAD (kg)	5950 GTO	6290 GTO	22,000 LEO

Ariane 5 - Vulcain Engine

* Major Engine Development for SEP and Other Subcontractors
* System Uses Gas Generator; Separate LH_2 and LOX Turbopumps
* Parameters:

Vacuum Thrust	1025 kN (230,000 lb)
Propellant	LH_2/LOX
O/F	5.2
I_{sp} (Vac)	431.6 s
P_c	100 Bars (1450 psi)
Engine Mass	1300 kg (2860 lbs)
LH2 Turbine Speed	34,200 rpm
LOX Turbine Speed	13,000 rpm
LH2 Turbine Power	11,300 kW
LOX Turbine Speed	2,900 kW

Ariane 5 - Solid Propellant Boosters (P 230)



* Being Led by Europulsion
* Development Plan Includes 10 Full-Scale Tests
* Uses Steel Case and HTPB/AP Propellants
* Known Motor Parameters:

Thrust	1,346,000 lb (5987 kN)
Propellant	HTPB/AP/AL
Propellant Mass	230,000 kg
Motor Diameter	3.05 m
Motor Length	30.6 m
Dry Mass	39,400 kg
t_b	125 s

Hermes

* Europe's Answer to Manned Spaceflight Independence; First Flight on Ariane 5 Currently Scheduled for 1998; Flight Rate of 2 Per Year
* Missions to: Columbus Free-Flying Lab, Space Station Freedom, Soviet Space Station Mir
* 3-Manned Crew, Delta-Winged Space Plane that Lands on Runway
* Hermes Consists of: Space Plane (13 m, 15 MT); Resource Module (6 m, 8 MT); and Propulsion Module (1 m); Hermes Robotic Arm (HERA)
* Storable Propellant Propulsion Module Consists of ACS with 32 - 20 & 400 N Class Storable Propellant Engines and 2 Main Orbit Injection Engines with 27.5 kN Thrust Level Each; Four Tanks Holding 7200 kg Propellant
* 3 MT Payload Capability; 2000 km Cross Range Landing Capability
* Includes a Crew Ejection System
* Primary Structure Made of Carbon-Resin Composites
* ~$5 Billion Program; ~170 Organizations with ~1500 People Working Presently, With ~5000 by 1992

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Hermes (Cont.)

* Hermes Participants: France 43.5%; Germany 27%; Italy 12.1%; Belgium 5.8%; Spain 4.5%; Netherlands 2.2%; Switzerland 1.5%; Sweden 1.3%, Canada 0.45%; Austria 0.5%; Denmark 0.45% and Norway 0.2%

* Phase B to be Complete Mid-1991; Program Phase C/D Expected to Begin in Late 1991

* Four Major Technology Challenges Identified by ESA:

 - Materials Necessary for Structures and Their Thermal Protection

 - Math Models for Aerodynamic, Aerothermal and Flight Simulations

 - H/O Fuel Cells for Electric Power

 - Flight Electronics and Software

Germany/MBB - Sanger Aerospace Plane

* Manned Reusable 2-Stage Winged HTO Vehicle Concept

* GLOW 340 MT; Airbreathing to M = 6.8 at 31 km; Uses Airbreathing LH_2 Turboramjet

* Aircraft Version Can Cruise at M = 4.4, and Carry 230 Passengers From Frankfurt to LA in Less Than 3 Hours

* Nominal Takeoff Thrust Level 1500 kN with 5 Engines

* Employs Expendable Stage CARGUS (Cargo Upper Stage) for 15 MT Payloads; From Ariane 5 Core Stage; Engine (HM.60) Thrust 1050 kN; LOX/LH_2

* Reusable Stage HORUS (Hypersonic Orbital Reusable Upper Stage) for Manned Missions; Main Engine Thrust 1200 kN; Expansion Ratio 325; I_{sp} = 472 s; O/F = 6.7; OMS Thrust 80 kN; I_{sp} = 437 s; Payload = 3300 kg

* MBB Is Conducting Technology Work on the Turboramjet for the First Stage of the Sanger; MBB Testing a GH_2 Ramjet Prototype in Mach 5 Wind Tunnel

* Funded at a $225 M Level Through 1992

* Will Likely Be a $10 B Demonstrator Program Funded by ESA

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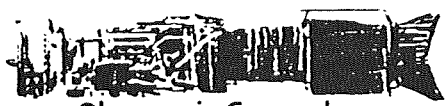
United Kingdom/British Aerospace - Hotol Aerospace Plane

* Horizontal Take Off and Horizontal Landing -- Hotol
* Manned Reusable Single-Stage-to-Orbit Vehicle Concept
* Uses Launch Trolley; GLOW 250 MT; Airbreathing to M=5 at 85 kft
* Employs Hybrid RB545 (Remains Classified) - Dual Mode Rocket Chamber that Utilizes Air as Oxidant in Lower Atmosphere
* H/O OMS; GH_2 RCS Thrusters
* Metal or Composite TPS, No Tiles
* Deliver 7 to 8 MT Into LEO
* Operational Cost ~ $4 to 5 M per Flight
* Proof of Concept in 1988; Wind Tunnel Tests from M = 5 to 18



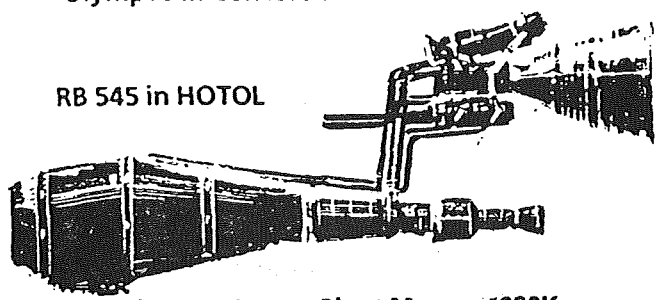
Hotol Propulsion System

Propulsion System Comparison



Olympus in Concorde

RB 545 in HOTOL



Olympus Power Plant Mass = 4080Kg

RB545 Power Plant Mass = 2840Kg

France - Star H Aerospace Plane

* French CNES Supported Study by Dassault/SNECMA/SEP/ONERA
* A Two-Stage to Orbit Aerospace Plane
* Reusable First Stage With Air-Breathing Engines
* An Expendable Second Stage With a HM-60 Cryogenic Engine
* Reusable Orbiter Derived from Hermes
* 400 MT GLOW; 280 MT 1st Stage; 120 MT 2nd Stage
* Studying Various Engine Cycles; Testing Scramjet; Wind Tunnel Tests

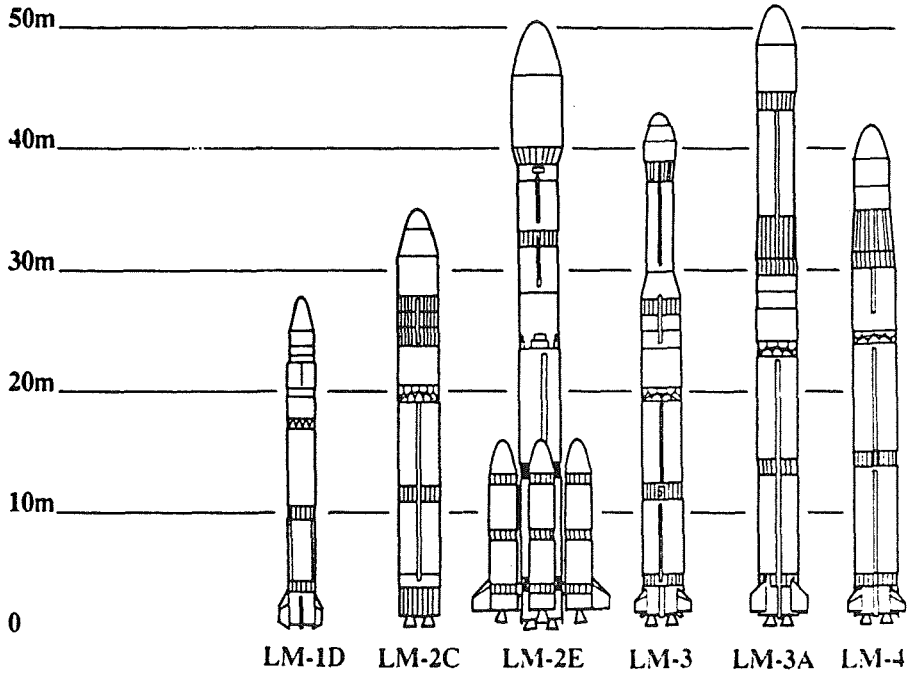
China - Long March Family

* China's Space Activities Date Back to the Late 1950's
* In 1964, China Launched It's First Launch Vehicle
* On April 24, 1970, China Launched it's First Earth Orbiting Spacecraft with Long March-1 (LM-1)
* In November 1975, China Launched a Recoverable Satellite Using LM-2
* In April 1984, LM-3 Was Successfully Launched
* China Has Now Developed a Successful, Reliable and Significantly Competitive Launch Capability

Long March Family of Launch Vehicles

Model	LM-1D	LM-2C	LM-2E	LM-3	LM-3A	LM-4
Overall length(m)	28	35	51	43.85	52.3	42
Lift-off weight(t)	80	191	464	202	240	249
Lift-off thrust(t)	112	284	600	284	300	300
Payload capability in LEO (kg)	700-750	2500	8800			2,500 (SSO)*
Payload capability in geostationary transfer orbit (kg)				1,400	2,500	
Available for commercial service	1991	1982	1990	1986	1992	1988

**SSO: Sunsynchronous orbit*



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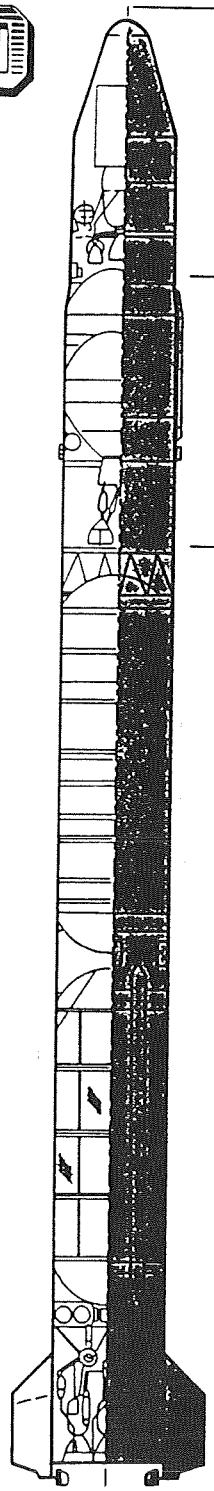
China's Launch History

		Launch Date	Launch Vehicle
1	Dong Fang Hong 1	24-4-1970	LM-1
2	Shi Jian 1	3-3-1971	LM-1
3	Technical Experiment	26-7-1975	LM-2A
4	Recoverable	26-11-1975	LM-2C
5	Technical Experiment	16-12-1975	FB-1
6	Technical Experiment	30-8-1976	FB-1
7	Recoverable	7-12-1976	LM-2C
8	Recoverable	26-1-1978	LM-2C
9	Shi Jian 2	20-9-1981	FB-1
10	Shi Jian 2A		
11	Shi Jian 2B		
12	Recoverable	9-9-1982	LM-2C
13	Recoverable	19-8-1983	LM-2C
14	Experimental	29-1-1984	LM-3
15	Experimental Geostationary Communications	8-4-1984	LM-3
16	Recoverable	12-9-1984	LM-2C
17	Recoverable	18-9-1985	LM-2C
18	Operational Geostationary Communications	1-2-1986	LM-3
19	Recoverable	6-10-1986	LM-2C
20	Recoverable	5-8-1987	LM-2C
21	Recoverable	9-9-1987	LM-2C
22	Operational Geostationary Communications	7-3-1988	LM-3
23	Recoverable	5-8-1988	LM-2C
24	Meteorological	7-9-1988	LM-4
25	Operational Geostationary Communications	22-12-1988	LM-3

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Long March - 1D

LM - 1D



Fairing

maximum diameter	2.054m
static effective diameter	1.754m

Third Stage

propellant (solid)	0.625t
diameter	2.05m

Second Stage

propellant ($UDMH/N_2O_4$)	12.2t
diameter	2.25m

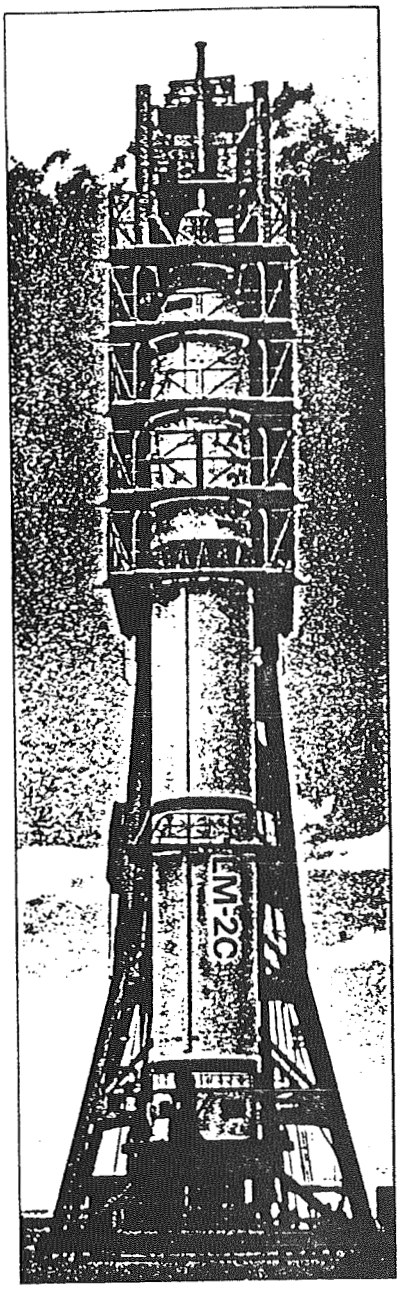
First Stage

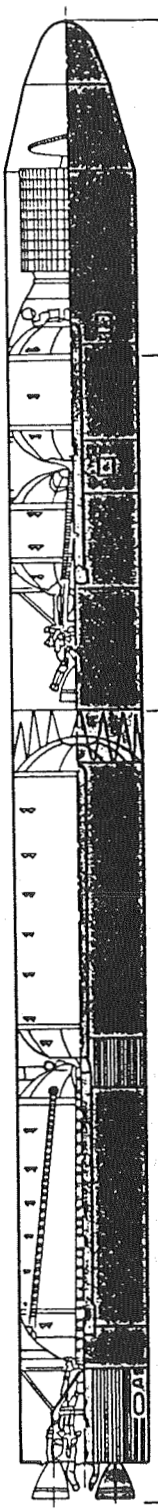
propellant ($UDMH/HNO_3$-27S)	60t
diameter	2.25m

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Long March - 2C

LM-2C





Fairing

maximum external diameter	3.35m
static effective diameter	3.07m

Second Stage

propellant ($UDMH/N_2O_4$)	35t
diameter	3.35m

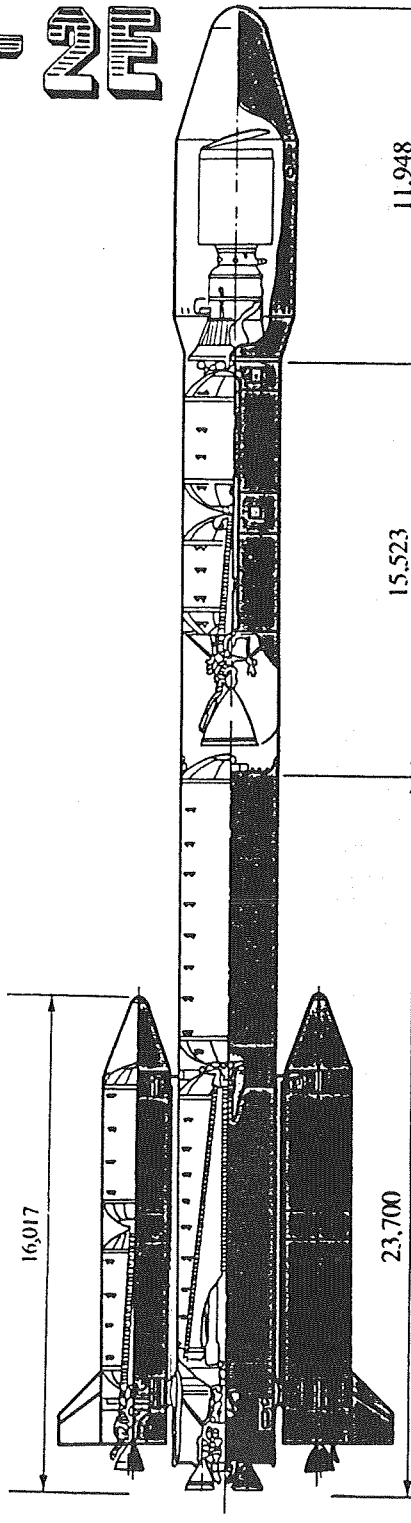
First Stage

propellant ($UDMH/N_2O_4$)	144t
diameter	3.35m

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Long March - 2E

LM-2E



Fairing

maximum external diameter	4.2m
static effective diameter	3.8m

Second Stage

propellant ($UDMH/N_2O_4$)	86t
diameter	3.35m

First Stage

propellant ($UDMH/N_2O_4$)	187t
diameter	3.35m

Liquid Strap-on Boosters

propellant ($UDMH/N_2O_4$)	4x37.5t
diameter	2.25m

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Long March - 3

LM-3



	(A)	(B)
Fairing		
maximum external diameter	2.60m	3.00m
static effective diameter	2.32m	2.72m
Third Stage		
propellant (LH/LOX)		8.5t
diameter		2.25m
Second Stage		
propellant ($UDMH/N_2O_4$)		35t
diameter		3.35m
First Stage		
propellant ($UDMH/N_2O_4$)		142t
diameter		3.35m

Long March - 3A

LM-3A



8,887

Fairing

maximum external diameter	3.35m
static effective diameter	3.00m

8,835

Third Stage

propellant (LH/LOX)	17.6t
diameter	3.00m

11,526

Second Stage

propellant ($UDMH/N_2O_4$)	29.6t
diameter	3.35m

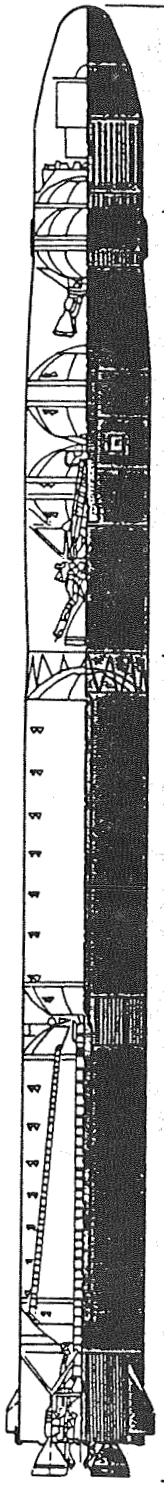
23,075

First Stage

propellant ($UDMH/N_2O_4$)	170t
diameter	3.35m

Long March - 4

LM-4



4,908

1,920

10,407

24,660

Fairing

maximum external diameter	2.90m
static effective diameter	2.36m

Third Stage

propellant ($UDMH/N_2O_4$)	14.15t
diameter	2.90m

Second Stage

propellant ($UDMH/N_2O_4$)	35.55t
diameter	3.35m

First Stage

propellant ($UDMH/N_2O_4$)	183.20t
diameter	3.35m

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India's Launch Vehicle Systems

* Satellite Launch Vehicle (SLV-3)
 - Approximately 15 MT Liftoff Mass
 - 23 m Long; 1 m Diameter Base
 - 4 Solid Propellant Stages; Segmenting Used on 1st Stage
 - 40 kg Payload Up to 800 km Circular at 45°
 - 4 Launches, 3 Successes in 1980 to 1983

* Augmented Satellite Launch Vehicle (ASLV)
 - SLV with 2 Solid Propellant Strapon Boosters
 - 150 kg Payload to LEO
 - 2 Failed Launches in 1987 and 1988
 - 2 Launched Scheduled Through 1993

* Polar Satellite Launch Vehicle (PSLV)
 - Approximately 275 MT Liftoff Mass
 - 44 m Long; 2.8 m Diameter Base
 - 4 Solid Propellant Stages; Solid/Liquid/Solid/Liquid
 - 1st Stage Has 6 Solid Motor Strapons
 - 1000 kg Payload Up to 900 km Circular at 90-100°
 - 3 Launches Scheduled Through 1994

* Geosynchronous Launch Vehicle (GSLV)
 - In Phase A/B; Goal Is 2 MT to GTO; Launches Planned in 1993-1995
 - Use Existing/Improved Stages, Plus New H/O Stage 3

India Propulsion Technology - PSLV Focused

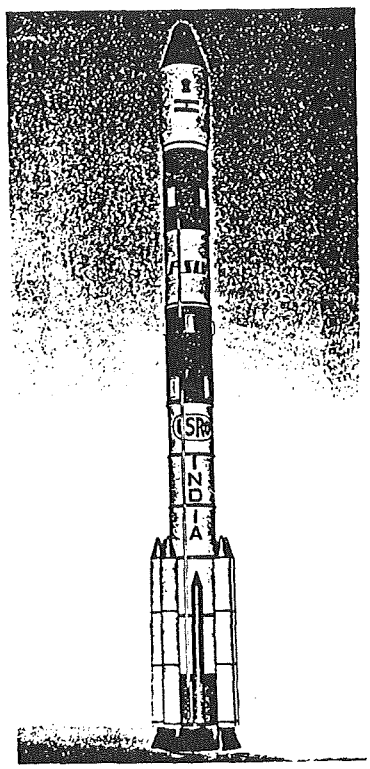
* 1st Stage (5 Segments) Is the 3rd Largest in the World
 - Motor Is 20 m Long and 2.8 m in Diameter
 - Uses HTPB-Based Propellant
 - Secondary Injection TVC Uses Strontium Perchlorate
 - Steel Case; Silicon Carbide Phenolic Composite Nozzle Liner
 - Successfully Static Tested on 10/21/89

* 2nd Stage Liquid
 - Viking Engine Licensed from SEP
 - UDMH/NTO Propellants
 - 8 Tests with 820 s Firing Time Completed

* 3rd Stage Solid
 - Motor is 2 m Long and 2 m in Diameter
 - Uses HTPB-Based Propellant
 - Submerged, Flex Seal Nozzle System
 - Kevlar Case
 - 2 Static Test Firings Completed in 4/89 and 1/90

* 4th Stage Liquid
 - Restart Capability
 - Engines Gimballed for TVC
 - Ti/Al Tankage and Structure
 - "Battleship" (Steel) Version Tested 7/89

India PSLV Model



Other Launch Vehicles

* Italy - Advanced Small Launch Vehicle (ASLV)
 - Fiat (Italy) and LTV (US) Formed Partnership in 11/89
 - Fait's SNIA BPD Subsidiary to Build Scout 2
 - Scout 2 is Scout with 2 Strapon Boosters
 - SNIA BPD to Market Europe; LTV North America
 - Launch from San Marcos, Wallops, Vandenberg
 - 460 kg Payload to 555 km Circular (San Marcos)
 - 4 Strapon Configuration Planned

* Israel - Shavit
 - Derived from Jericho Missile
 - 2 Successful Launches, 9/88 and 4/90
 - 160 kg Payload to 210 x 1500 km at 143°
 - 3-Stage, Solid Propellant

* Iraq Has a Launcher Called ABID, a 3-Stage Missile System

* South Korea Has Announced Plans to Build Satellites and Develop an Independent Launch Capability

* Norway is Developing LittLEO, a Scout-Class Launcher; 1st Launch in 1991

* Brazil Just Cancelled It's Program by the New Government

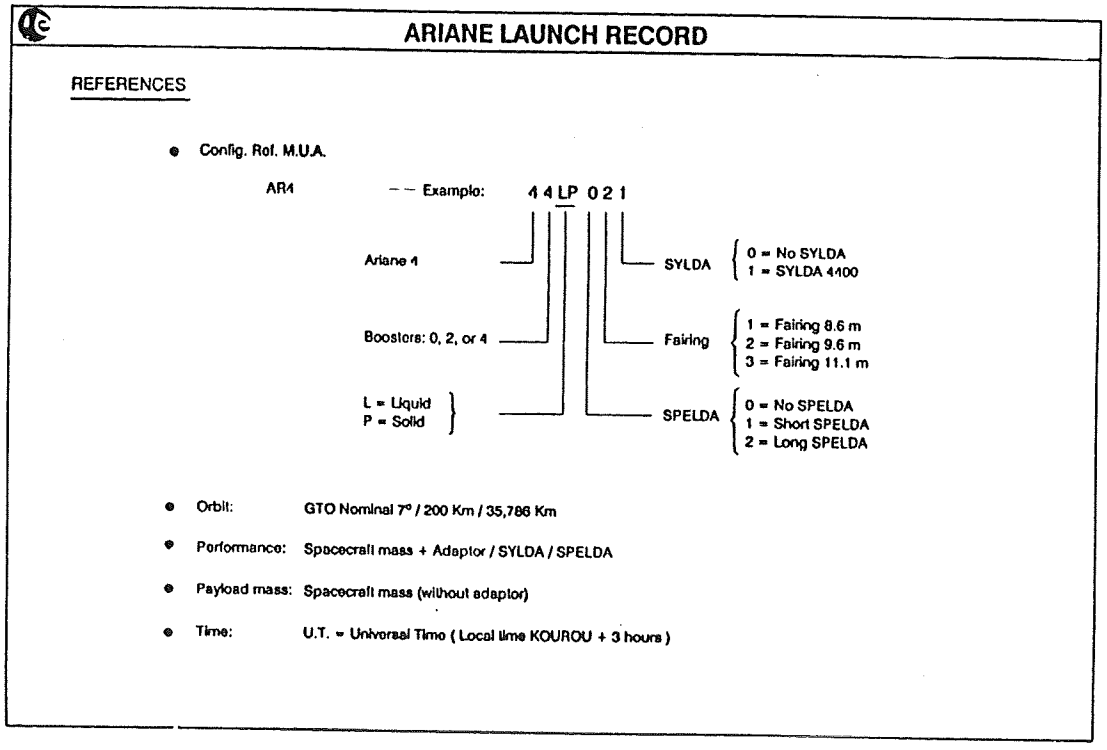
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Ariane Launch Record

Issue: V36

by

European Space Agency



ARIANE LAUNCH RECORD

LAUNCHER				IGNITION		PAYLOAD				ESA REPORT	REMARKS
FLIGHT N°	CONFIG. Ref. M.U.A.	PERFORM Kg	ORBIT	DATE	TIME U.T.	CUSTOMER / MANUFACTURER	NAME	MASS Kg	MISSION / FINAL ORBIT	ESA/PB-AR	
L01	AR1 ACU 1194	1645 (48.2)	GTO 17.5°	24-12-79	17h14'38"	ESA / AEROSPATIALE	CAT / BALLAST	1602	TECHNOLOGICAL / GTO	(80)1 (80)8 (80)17	1st ARIANE 1 1st QUALIFICATION FLIGHT
L02	AR1 ACU 1194	1643	[GTO 17.5°]	23-05-80	14h28'39"	MAX PLANK INST. / MBB AMSAT CORP. / AMSAT ESA / AEROSPATIALE	FIREWHEEL OSCAR 9 CAT	1080 82 340	SCIENTIFIC / [GTO] RADIO AMATEURS / [ELLIPTIC] TECHNOLOGICAL / [GTO]	(80)21 (80)29 (80)40	2nd QUALIFICATION FLIGHT FAILURE OF FIRST STAGE
L03	AR1 ACU 1194	1678	GTO 10.5°	19-06-81	12h12'59"	ESA / AEROSPATIALE INDIAN SPACE RESEARCH ORG. / ISRO ESA / AEROSPATIALE	METEOSAT APPLE CAT	717 630 140	METEOROLOGICAL / GEO COMMs & TECHNO. / GEO TECHNOLOGICAL / GTO	(81)28	3rd QUALIFICATION FLIGHT
L04	AR1 ACU 1194	1699	GTO	20-12-81	01h29'00"	ESA / BAe ESA / MATRA ESA / AEROSPATIALE	MARECS - A VID CAT & (THESEE)	995 503 158	MARITIME COMMs / GEO TECHNOLOGICAL / GTO TECHNO. & (SCIENTIF.) / GTO	(82)1	4th QUALIFICATION FLIGHT
L5	AR1 / SYLDA 3900	1617 (184)	[GTO 8°]	10-09-82	02h12'00"	ESA / BAe ESA / CNI	MARECS-B SIRIO 2	1014 419.5	MARITIME COMMs / [GEO] COMMs & TIME SYNCHRO. / [GEO]	(82)31	FAILURE OF THIRD STAGE
L6	AR1 / SYLDA 3900	1341	GTO 8.6°	16-06-83	11h59'03"	ESA / BAe AMSAT CORP. / AMSAT	ECS-1 AMSAT Ph IIIB	1043 130	COMMUNICATIONS / GEO RADIO AMATEURS / ELLIPTIC.	(83)30	COLLISION 3rd STAGE / AMSAT AFTER SEPARATION
L7	AR1 Attach fitting ACU 1497	1861 (21) (27)	GTO 8.5° 185Km	19-10-83	00h45'36"	INTELSAT / FORD	INTELSAT V / F7	1813	COMMUNICATIONS / GEO	(83)46	
L8	AR1 Attach fitting ACU 1497	1881 (21) (27)	GTO 8.5° 185Km	05-03-84	00h50'03"	INTELSAT / FORD	INTELSAT V / F8	1833	COMMUNICATIONS / GEO	(84)13	

ALL LAUNCHES FROM ELA 1

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ARIANE LAUNCH RECORD

LAUNCHER				IGNITION		PAYLOAD				ESA REPORT	REMARKS
FLIGHT N°	CONFIG. Ref. M.U.A.	PERFORM Kg	ORBIT	DATE	TIME U.T.	CUSTOMER / MANUFACTURER	NAME	MASS Kg	MISSION / FINAL ORBIT	ESA/PB-AR	
V9	AR1 ACU 937	1237 (47)	GTO	23-05-84	01h33'29"	GTE / RCA	SPACENET F1	1195	COMMUNICATIONS / GEO	/MIN/58	1st COMMERCIAL FLIGHT
V10	AR3 SYLDA 4400 Att. Ring TC1	2576 (12) (208)	GTO	04-08-84	13h32'54"	ESA / BAe FRENCH PTT / MATRA	ECS-2 TELECOM 1A	1172 1196	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(84)46	1st ARIANE 3
V11	AR3 SYLDA 4400 SH.	2445 (197.4) (68.4)	GTO	10-11-84	01h14'15"	ESA / BAe GTE / RCA	MARECS-B2 SPACENET F2	1060 1196	MARITIME COMMs / GEO COMMUNICATIONS / GEO	(84)61	
V12	AR3 SYLDA 4400	2552	GTO	08-02-85	23h22'00"	ARABSAT SATEL. COMM. ORG. / AEROSPATIALE EMBRATEL / SPAR	ARABSAT F1 BRASILSAT F1	1215 1140	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(85)8	
V13	AR3 SYLDA 4400 Att. Ring TC1	2645.4 (204.5)	GTO	08-05-85	01h15'38"	GTE / RCA FRENCH PTT / MATRA	G-STAR 1 TELECOM 1B	1230.4 1210.5	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(85)30	
V14	AR1 ACU 937	1005.5 (48)	GTO	02-07-85	11h23'13"	ESA / BAe	GIOTTO	957.5	SCIENTIFIC / HELIOCENTRIC	(85)37	ATTEMPT RECOVERY 1st STAGE FAILED
V15	AR3 SYLDA 4400	2565.1 (195.6)	[GTO]	12-09-85	23h26'00"	GTE / RCA ESA / BAe	SPACENET F3 ECS-3	1195.8 1173.7	COMMUNICATIONS / [GEO] COMMUNICATIONS / [GEO]	(85)54	FAILURE 3rd STAGE
V16	AR1 ACU 1194	2422 (49)	SSO 98.4° 836.5 / 816.5 Km	22-02-86	01h44'35"	CNES / MATRA SWEDISH SPACE CORP. / SAAB	SPOT-1 VIKING	1835 538	EARTH OBSERVATION / SUNSYNCHRONOUS SCIENTIFIC / POLAR, 600-15,000 Km	(86)11	LAST ARIANE 1 1st NORTHWARD LAUNCH

ALL LAUNCHES FROM ELA 1

ARIANE LAUNCH RECORD

LAUNCHER				IGNITION		PAYLOAD				ESA REPORT	REMARKS
FLIGHT N°	CONFIG. Ref. M.U.A.	PERFORM. Kg	ORBIT	DATE	TIME U.T.	CUSTOMER / MANUFACTURER	NAME	MASS Kg	MISSION / FINAL ORBIT	ESA/PB AR	
V17 ☆	AR3 SYLDA 4400	2631.1 (193.7)	GTO	28-03-86	23h30'00"	GTE / RCA EMBRATEL / SPAR	G-STAR 2 BRASILSAT F2	1242.6 1194.8	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(86)18	1st LAUNCH FROM ELA 2
V18	AR2 ACU 1497 Alt. filling	2004.3 (27) (20)	[GTO 8° 463 Km]	31-05-86	00h53'00"	INTELSAT / FORD	INTELSAT V F14	1956.5	COMMUNICATIONS / [GEO]	(86)30	1st ARIANE 2 FAILURE OF 3rd STAGE
V19	AR3 SYLDA 4400	2562.5 (193.5)	GTO	16-09-87	00h45'28"	AUSSAT Pty Ltd / HUGHES ESA / BAe	AUSSAT K3 ECS-4	1195.6 1173.4	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(87)60	
V20 ☆	AR2 Specific ACU	2134 (53)	GTO 4°	21-11-87	02h19'00"	MIN. FOR RESEARCH AND TECH IN. OF RFD / EUROSATELLITE	TV SAT 1	2081	COMMUNICATIONS / GEO	(87)70	
V21	AR3 SYLDA 4400 TCI ACU ring Clampband	2632.8 (190.9) (4.5) (6.6)	GTO	11-03-88	23h28'00"	SPACENET / RCA FRANCE TELECOM / MATRA	SPACENET IIIC / GEOSTAR R01 TELECOM 1C	1216.8 1214	COMMUNICATIONS / GEO COMMUNICATIONS / GEO	(88)17	
V23	AR2 ACU 1497 Alt. Ring	2024 (27) (21)	GTO 8° 463 Km	17-05-88	23h58'00"	INTELSAT / FORD	INTELSAT V F13	1978	COMMUNICATIONS / GEO	(88)32	
A401 (V22) ☆	AR44LP 120 SPELDA (SP. H.) Carr. struct. ACU 937 Ballast	3497 (409.7) (193.8) (767.5) (47.35) (222)	GTO 10° 220 Km	15-06-88	11h19'01"	ESA / AEROSPATIALE AMSAT CORP. / AMSAT PAN AMERICAN SATELLITE / RCA	METEOSAT P2 OSCAR 13 PAN AMERICAN SATELLITE	696 144 1220	METEO & TIMESYNCHRO / GEO RADIO AMATEURS / ELLIPTIC COMMUNICATIONS / GEO	(88)33	1st ARIANE 4 DEMONSTRATION FLIGHT
V24	AR3 SYLDA 4400	2565.9 (189.4)	GTO	21-07-88	23h12'00"	INDIAN SPACE RESEARC ORG. / ISRO ESA / BAe	INSAT 1C ECS-5	1191.1 1185.4	COMMs & METEO / GEO COMMUNICATIONS / GEO	(88)39	

☆ LAUNCHES FROM ELA 2

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ARIANE LAUNCH RECORD

LAUNCHER				IGNITION		PAYLOAD				ESA REPORT	REMARKS
FLIGHT N°	CONFIG. Ref. M.U.A.	PERFORM Kg	ORBIT	DATE	TIME U.T.	CUSTOMER / MANUFACTURER	NAME	MASS Kg	MISSION / FINAL ORBIT	ESA/PB-AR	
V25 *	AR3 SYLDA 4400	2699.8 (189.3)	GTO	08-09-88	23h00'00"	GTE / RCA	G-STAR III / GEOSTAR R02	1271.5	COMMUNICATIONS / GEO	(88)51	
						SAT. TRANSPONDER LEASING / HUGHES	SBS 5	1239	COMMUNICATIONS / GEO		
V26	AR2 Specific ACU	2130.7 (54)	GTO 4° 265 Km	26-10-88	02h17'03"	TELEDIFFUSION DE FRANCE / EUROSATELLITE	TDF 1	2076.7	TV BROADCASTING / GEO	(88)66	
V27 *	AR44LP 120 ACU 937 SPELDA ACU 1194	3710 (49.4) (417) (43.4)	GTO	11-12-88	00h33'38"	UK MINISTRY OF DEFENCE / BAe	SKYNET 4B	1429.1	MILITARY COMMs / GEO	(89)1	
						SOCIETE EUROP.DES SATELLITES / RCA	ASTRA 1	1767.7	TV BROADCASTING / GEO		
V28	AR2 ACU 1497 Attach fitting	2027 (26) (20)	GTO 8° 463 Km	27-01-89	01h21'00"	INTELSAT / FORD	INTELSAT V F15	1981	COMMUNICATIONS / GEO	(89)2	
V29 *	AR44LP 120 ACU 1666A SPELDA ACU 937C	3475.1 (48.7) (412) (52.6)	GTO	06-03-89	23h29'00"	JAPAN COMM. SAT. COMPANY / HUGHES	JC-SAT 1	2280	COMMUNICATIONS / GEO	(89)16	
						ESA / AEROSPATIALE	MOP 1	681.5	METEOROLOGICAL / GEO		
V30	AR2 Specific ACU	2142 (52)	GTO 4° 250 Km	02-04-89	02h28'00"	NORDIC SAT. CPNY / EUROSATELLITE	TELE-X	2090	TV BROADCASTING / GEO	(89)25	LAST ARIANE 2
V31 *	AR44L 110 ACU 1194B SPELDA ACU 937	4418 (62.7) (405.4) (48.4)	GTO	05-06-89	22h37'18"	SPACE COMM. CORP / FORD	SUPERBIRD A	2469.2	COMMUNICATIONS / GEO	(89)36	FIRST AR44L / UPGRADED PERFORMANCE (213 Kg ABOVE SPECIFIED MAXIMUM)
						DEUTSCHE BUNDES POST / KONSORTIUM R-DFS	DFS KOPERNIKUS	1415.8	COMMUNICATIONS / GEO		
V32	AR3 ACU 1194	2660.3 (48.3)	GTO	12-07-89	00h14'00"	ESA / BAe	OLYMPUS	2612	COMMUNICATIONS / GEO	(89)37	LAST ARIANE 3 LAST LAUNCH FROM ELA 1

* LAUNCHES FROM ELA 2

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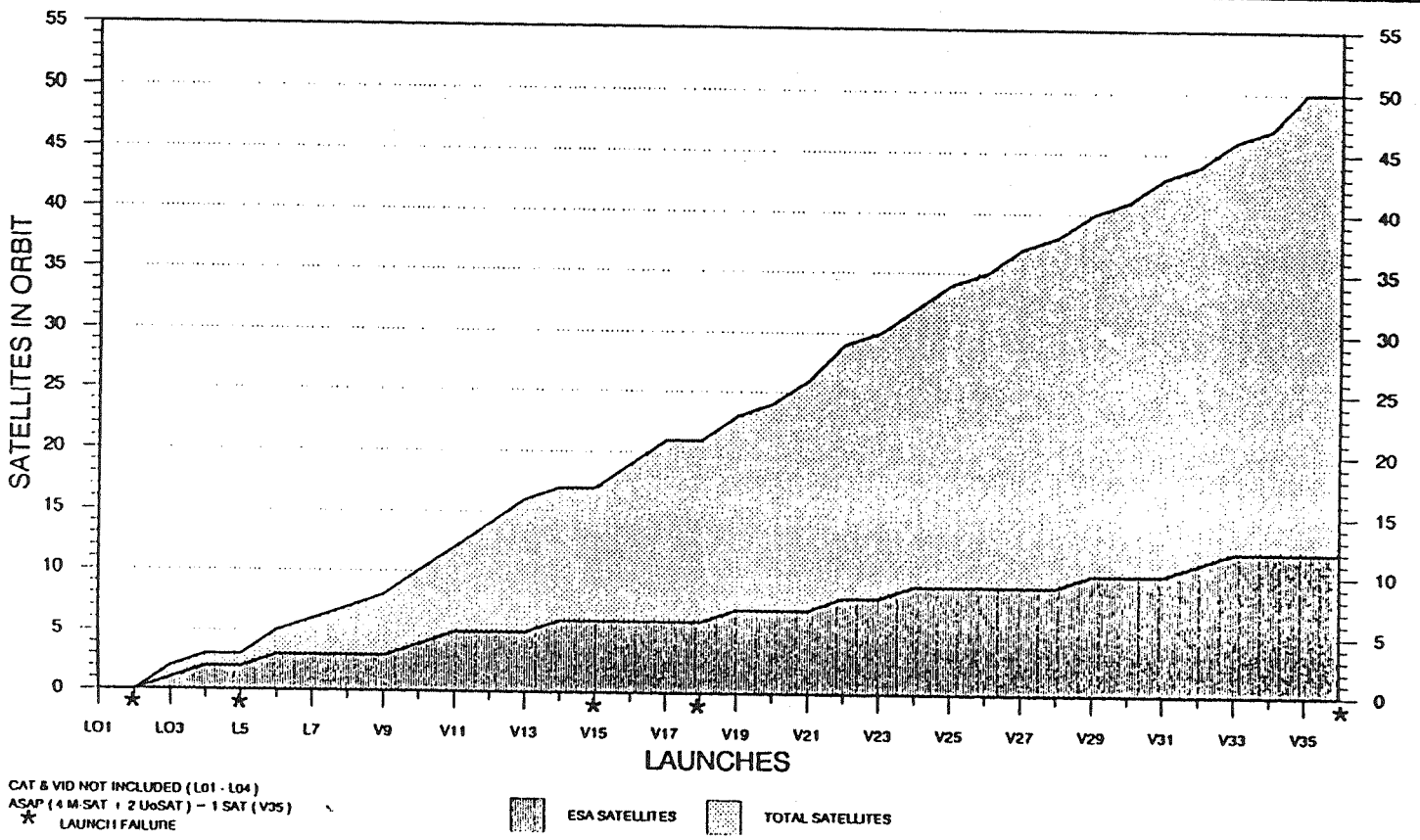
ARIANE LAUNCH RECORD

LAUNCHER				IGNITION		PAYLOAD				ESA REPORT	REMARKS
FLIGHT N°	CONFIG. Ref. M.U.A.	PERFORM Kg	ORBIT	DATE	TIME U.T.	CUSTOMER / MANUFACTURER	NAME	MASS Kg	MISSION / FINAL ORBIT	ESA/PB-AR	
V33	AR44LP 120 Specific ACU SPELDA ACU 937	3730 (53) (417.6) (47.4)	GTO	08-08-89	23h25'53"	MIN. FOR RESEARCH AND TECHN. OF FRG / EUROSATELLITE	TV SAT 2	2073	COMMUNICATIONS / GEO	(89)38	
						ESA / MATRA	HIPPARCOS	1139	ASTRONOMETRY MISSION / GTO		
V34	AR44L 020 Specific ACU	4286 (75)	GTO	27-10-89	23h05'00"	INTELSAT / HUGHES	INTELSAT VI F2	4211	COMMUNICATIONS / GEO	(89)59	
V35	AR40 020 ACU 1194 BALLAST A.S.A.P.	2215.2 (57.4) (81.2) (60.65)	SSO	22-01-90	01h35'27"	CNES / MATRA	SPOT-2	1875	EARTH OBSERVATION / SUNSYNCHRONOUS	(90)4	
						UoSAT/SST / UNIVERSITY OF SURREY - UK	UoSAT D	46	EXP. SCIENTIFIC, TELECOM, TECHNO. DEMO. / POLAR		
							UoSAT E	47			
						AMSAT N.A. - USA	Microsat A	13.34	EDUCATION AND COMMUNICATION / POLAR		
							Microsat B	12.92			
							Microsat C	16.03			
							Microsat D	13.76			
V36	AR44L 120 ACU 1194B SPELDA ACU 937	4250 (49) (403.3) (46.5)	[GTO]	22-02-90	23h17'00"	Space Communication Corporation (SCC) / FORD	SUPERBIRD - B	2501.3	COMMUNICATIONS / [GEO]	(90)12	FAILURE OF FIRST STAGE
						Nippon Hodo Kiyokai (NHK) / GE Astro Space Division	BS - 2X	1249.9	COMMUNICATIONS / [GEO]		

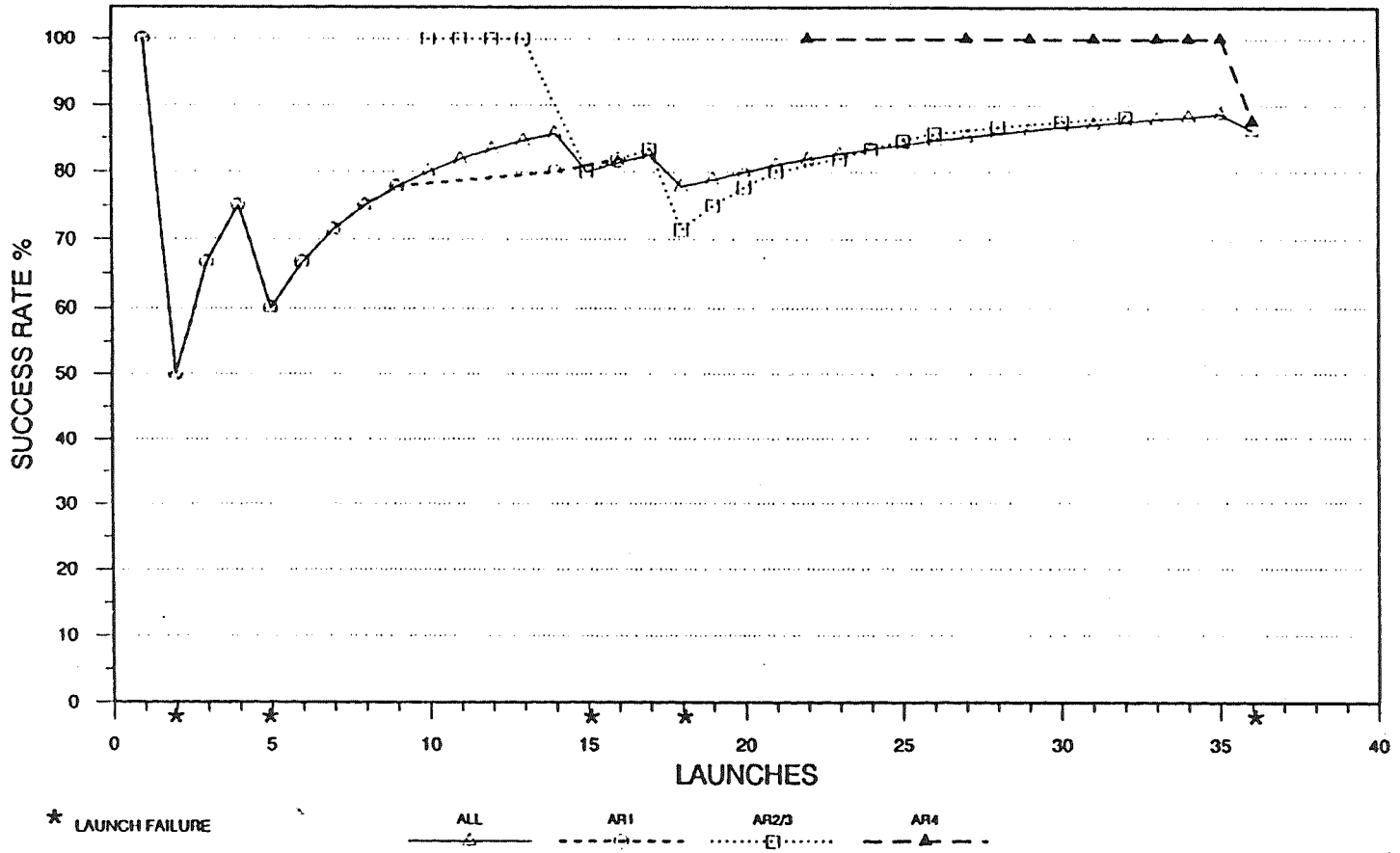
ALL LAUNCHES FROM ELA 2

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ARIANE LAUNCH RECORD



ARIANE LAUNCH RECORD



ENVIRONMENTAL CONSIDERATIONS

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Joyce Jatko NASA HQ

ENVIRONMENTAL CONCERNS IN PROPULSION TECHNOLOGY

ENVIRONMENTAL MEDIA

- AIR EMISSIONS
 - RESTRICTIONS DEPENDENT ON LOCATION OF TESTING
 - REGULATED BY TOTAL EMISSIONS OR CHANGE TO AMBIENT AIR QUALITY
 - CONTROL TECHNOLOGY NOT ALWAYS AVAILABLE (DEVELOPMENTAL REQMT)
 - CHLOROFLUOROCARBONS (CFCs) - RESTRICTIONS ON AVAILABILITY

- NUCLEAR
 - PUBLIC CONCERN OVER SAFETY
 - DOE PROBLEMS AND RESULTANT PUBLIC PERCEPTIONS

- HAZARDOUS WASTE MANAGEMENT
 - REGULATIONS BECOMING MORE RESTRICTIVE
 - EXOTIC FUELS MORE DIFFICULT TO DISPOSE OF
 - DISPOSAL COSTS ACCELERATING
 - WASTE PROPELLANT DISPOSAL OPTIONS BECOMING MORE LIMITED

- NATIONAL ENVIRONMENTAL POLICY ACT (NEPA)
 - NEPA CONCERNS MUCH GREATER VISIBILITY THAN IN PAST
 - PROCESS REQUIRED PRIOR TO IMPLEMENTING NEW PROGRAMS
 - 12-18 MONTH PROCEDURE BEFORE RECORD OF DECISION CAN BE ISSUED

IMPACT TO PROGRAM

- SCHEDULE: NEED TO PLAN ON TIME FOR:
 - NEPA PROCESS (NEW TEST SITES)
 - PERMITS (AIR AND/OR WATER DISCHARGES, HAZARDOUS WASTE

- COST (INCREASED COSTS FOR TESTING PROGRAMS)
 - HAZARDOUS WASTE DISPOSAL COSTS INCREASING SHARPLY
 - CFC's INCREASING IN COST, DECREASING AVAILABILITY
 - FUNDING OF R&D EFFORT FOR:
 - CLEANER PROPELLANT
 - WASTE MINIMIZATION
 - MATERIALS SUBSTITUTION

- LOCATION OF TESTING
 - SOME SITES MAY REPRESENT INCREASED ENVIRONMENTAL COSTS
 - TESTING MAY NEED TO BE PERFORMED AT SITE ALREADY PERMITTED
 - THERE MAY BE RESTRICTIONS ON EXPANSION OF EXISTING FACILITIES

NEED FOR GREATER COOPERATION AMONG CENTERS

- SUPPORT FOR TEST PROGRAMS IN LESS ENVIRONMENTALLY SENSITIVE AREAS

- MORE SHARING OF TEST FACILITIES

- PLANNING FOR ENVIRONMENTAL COMPLIANCE
 - ADVANCED PLANNING AND COORDINATION
 - COST AND SCHEDULE IMPACTS

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NASA PROPULSION ENGINEERING RESEARCH CENTER AT PENN STATE SECOND ANNUAL SYMPOSIUM

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PENNSTATE



NASA Propulsion Engineering Research Center at Penn State

Second Annual Symposium

Thursday, June 28, 1990
Kern Auditorium
The Pennsylvania State University
University Park, PA

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AGENDA

Session A: Liquid Propellant Combustion
Rm. 112 Kern

Session Chairman: Dr. Robert J. Santoro

2:00	Dr. Charles L. Merkle, Director	Center Overview
2:30	Dr. Kenneth K. Kuo and Dr. Robert J. Santoro	Cryogenic Combustion Laboratory
3:00	Dr. Stephen R. Turns	Ignition and Combustion of Metallized Propellants
3:30	Dr. Vigor Yang	Theoretical Study of Combustion Instabilities in Liquid-Propellant Rocket Motors
4:00	Dr. Harold R. Jacobs and Dr. Robert J. Santoro	Spray Combustion under Oscillatory Pressure Conditions
4:30	Dr. Fan-Bill Cheung and Dr. Kenneth K. Kuo	Liquid Jet Breakup and Atomization in Rocket Chambers under Dense Spray Conditions with Compression/Shock Wave Interaction
5:00	Dr. Domenic Santavicca	Turbulence-Droplet Interactions in Vaporizing Sprays
		Laser Spark Ignition

Session B: Liquid Propulsion Technologies
Rm. 101 Kern

Session Chairman: Dr. Michael M. Micci

2:00	Dr. Charles L. Merkle, Director	Center Overview **(Rm. 112 Kern)**
2:30	Dr. Robert Pangborn and Dr. Richard A. Queeney	Hydrogen Management in Materials for High Pressure Hydrogen/Oxygen Engines
3:00	Dr. Alok Sinha and Dr. Kon-Well Wang	Robust and Real-Time Control of Magnetic Bearings for Advanced Propulsion Rockets
3:30	Dr. Marc Carpino	Analysis of Foil Bearings for High Speed Operation in Cryogenic Applications
4:00	Dr. Laura Pauley	A Study of Methods to Investigate Nozzle Boundary Layer Transition
4:30	Dr. Michael M. Micci	Optical Diagnostic Investigation of Low Reynolds Number Nozzle Flows
5:00	Dr. Charles L. Merkle	Flowfield Analysis of Low Reynolds Number Rocket Engines

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AN OVERVIEW OF THE PENN STATE PROPULSION ENGINEERING RESEARCH CENTER

AN OVERVIEW OFTHE PENN STATE PROPULSION ENGINEERING RESEARCH CENTER

Charles L. Merkle
Department of Mechanical Engineering

I. Research Objectives and Long Term Perspective of the Center

Penn State's Propulsion Engineering Research Center was established in August, 1988 under a grant from NASA's University Space Engineering Research Centers (USERC) Program. The Center includes participation from the Colleges of Engineering and Science at Penn State, and a cooperative program with Lincoln University. The Center's primary focus is to conduct research and educate students in the broad areas of space propulsion, but it also includes auxiliary efforts in gas turbine propulsion, internal combustion engines, and some topics in marine propulsion. There are currently fourteen faculty participating in the Center. In addition, some twenty-seven graduate students and sixteen undergraduate students are supported by the Center. The Center's research program is focussed around five concentrations: Combustion, Fluid Mechanics and Heat Transfer, Materials Compatibility, Turbomachinery, and Advanced Propulsion Concepts. Downstream plans include broadening the effort in turbomachinery and adding efforts on electric and nuclear propulsion.

The objectives of the Propulsion Center are to provide a focussed research effort in space propulsion that will attract students to space engineering opportunities and will provide a continuing supply of graduates at all degree levels with interest and expertise in space propulsion. A parallel objective is to enhance participation in engineering by women and under-represented minorities. As space exploration and development mature, space activities will have a larger and larger impact on world economics. The United States needs to ensure an adequate supply of engineers and scientists with expertise in these areas if we are to compete in this emerging world market. The Center's goal is to provide graduates for this expanding field, as well as to provide research advances that will lead to improved technologies.

The organizational structure of the center includes a Director, an external Policy Advisory Board, and an internal Faculty Advisory Board. The Director has responsibility for day-to-day operation of the Center and for ensuring that it works in an integrated fashion toward a common goal. The external Policy Advisory Board assists him in matters of policy and research emphasis, while the internal Faculty Review Board assists in decision-making. The Policy Advisory Board is composed of leaders from government, industry, and academia and is charged with guiding the long range development of the Center. The Policy Board has one formal and one informal meeting each year to evaluate Center progress, and to advise as to appropriate technical direction. The Faculty Review Board reviews internal proposals for research projects, including in their deliberations evaluations from members of the Policy Advisory Board. The Faculty Review Board is composed of senior faculty plus the Senior Vice-President for Research and Graduate Studies.

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The responsibilities for directing individual research projects are delegated to individual faculty in a mix of three categories of research programs: Core Research Projects, Matching Funds Projects and Exploratory Projects. The first two of these provide for multi-year, research programs. Core Projects are funded by the Center; Matching Projects receive shared funds from the Center and outside agencies. Exploratory projects are small, short-term efforts to establish feasibility of new ideas. The individual PI's work in close fashion with each other and the Director to provide the cross-fertilization that ensures that the whole of the Center's output is more than the sum of its parts.

In our first two years of operation, the Center has, indeed, had a major impact on graduate student enrollment and has enabled us to start a small, but highly successful, minority program. Graduate student involvement in the Center is through two paths. We offer NASA Traineeships which are funded through the Center itself, and Research Assistantships which are funded through the individual projects that comprise the Center. Of the 31 graduate students currently supported by the Center, 28 are U. S. citizens. Undergraduate involvement is fostered by a summer research program that is focused on minority students from both Lincoln University and Penn State University. This program included three minority students in our first year, and is supporting five students this summer. We also have Penn State undergraduates involved at the Center during the school year. During the recently completed academic year, we had two graduate and two undergraduate minority students working in the Center.

II. Current Status and Operational Philosophy

At the outset of the Center, we deliberately chose a start-up philosophy that focused attention toward a narrow facet of space propulsion. This allowed us to begin in an orderly fashion with a truly integrated "Center" concept while laying the foundation for later expansion into a more broadly based program. The choice for our initial focus was liquid rocket propulsion systems. This choice was made because of the dominance of liquid rocket engines in present and future space transportation programs of the United States, and because it was an area which had seen but little research emphasis in the previous ten or fifteen years.

To ensure a common thread of continuity in our initial research projects, we specialized even further in the first year by concentrating on combustion-related issues of liquid rockets including fluid dynamics, heat transfer and materials compatibility issues. Combustion and combustion devices represent an important subset of problems in liquid rocket propulsion, and, in addition, represent an area of strength at Penn State. At present this portion of the Center constitutes a fairly mature area; the projects are all well established and are providing significant research results. Downstream projections for the combustion area are to maintain it at about the present level augmenting in part the Center funds by auxiliary funds from other sources.

In the second year, we initiated an effort in the turbomachinery aspects of liquid propulsion engines as a first major step in broadening the focus of the Center. This second major thrust is just currently becoming established and is still growing in

scope. Additional efforts in turbomachinery-related areas are planned for the immediate future.

In addition to this major emphasis on liquid rocket propulsion, we have also maintained modest efforts on advanced propulsion concepts. The Center is currently providing some support for a research effort on antimatter propulsion which is primarily supported by JPL and AFAL. There are also auxiliary efforts on microwave propulsion and advanced electric concepts. We expect additional growth in these non-chemical propulsion programs in the coming years.

An important part of our Center is the development and use of a major new cryogenic laboratory with ultimate capability for liquid oxygen and liquid hydrogen or liquid hydrocarbon propellants. Detail design and construction of this laboratory was begun shortly after Center start-up and our first hot firing was made with gaseous propellants in December 1989. This unique laboratory is currently the only one of its kind in U. S. universities. Similar systems used in the Sixties have been mothballed or dismantled. Now that the facility is operational for gaseous propellants, we are beginning testing and diagnostics at appropriate conditions of interest. Evolution of the propellant capabilities to cryogenic liquids is continuing, with a target of demonstrating LOX capability in calendar year 1990. The Cryogenic Laboratory enables us to do small scale tests (generally with uni-element injectors) with actual propellants under realistic conditions. The laboratory also enables us to give students experience in handling the cryogenic fluids that are generally used in space propulsion applications. The construction of this laboratory would have been totally impossible without the Center.

In all the areas described above, there is an integrated treatment of experimental and analytical efforts with close interaction among both faculty and students. This interaction is facilitated by the co-location of all faculty and students in the Center and through the shared use of the new Cryogenic Laboratory which is just being brought on line. The Cryogenic Laboratory is to be used for both combustion and materials testing by several Center projects.

In January of 1989, the Center became the first occupants of the newly constructed Research Building . The space originally allocated to the Center was the first floor of this building, but as the Center became established, it first expanded to include one quarter of the second floor, and is now occupying the recently completed basement floor bringing our total assigned area to about 16,000 square feet. This building offers excellent laboratory space with adjacent offices and provides a common working area for Center personnel. At present, the Research Building houses 12 faculty members,11 staff and approximately 50 graduate students along with 9 laboratories. The Center is also assigned 1,000 square feet of space in the High Pressure Laboratory for our Cryogenic Combustion Laboratory. Safety precautions prevent housing this facility in the main Center location.

In addition to research and student support, the Center has also had an impact on our instruction program. Penn State already had a rich offering of courses in propulsion when the Center began, but in conjunction with the Center, we have developed two new graduate offerings in propulsion and a third is tentatively planned

for the 1990-91 academic year. The course instructors are Prof. Micci of Aerospace Engineering and Profs. Yang and Carpino of Mechanical Engineering.

III. Brief Description of Center Projects

The projects supported in the Center are divided into five concentrated areas. Each of these is outlined briefly below. More detail on the individual projects is given in the following papers in this volume.

Combustion Concentration: The combustion concentration was the first area established and remains the largest. Research thrusts in combustion include both experimental and analytical efforts and extend rather broadly across several propulsion areas. Much of the effort is directed towards the understanding of spray combustion phenomena. Specific experimental research includes studies of liquid jet break-up and atomization under both dense and dilute spray conditions. Non-obtrusive diagnostic procedures include optical techniques in the dilute spray regions and non-optical techniques in the dense spray regions. Particular optical diagnostics being used include laser doppler velocimetry, phase doppler particle anemometry, laser speckle velocimetry, advanced flame imaging techniques and planar laser-induced fluorescence. In non-optical diagnostics we are focusing on high intensity x-ray radiography.

Other aspects of spray combustion phenomena being studied include measurements of droplet-turbulence interactions and fundamental studies of the burning mechanisms of slurry fuel droplets. These experimental studies are corroborated by supporting theoretical research in multicomponent droplet vaporization in the supercritical and near-supercritical regimes as well as theoretical and numerical analyses of combustion instability phenomena.

A current focus of the combustion area is upon the construction of a major new Cryogenic Facility which will enable experimental studies of the combustion of hydrogen or liquid hydrocarbon fuels with cryogenic liquid oxygen. This facility provides us with an invaluable tool for studying the combustion processes in liquid rocket engines. The initial phase of this project was completed in calendar year 1989 and we are presently conducting initial tests with gaseous oxygen . The cryogenic (liquid) capability is expected to be completed in 1990.

Fluid Mechanics and Heat Transfer Concentration: The fluid mechanics and heat transfer concentration is directed towards experimental and computational studies of a broad range of propulsion applications. In the area of space propulsion, we are conducting experimental surveys of the boundary layers in low Reynolds number nozzles using emission spectroscopy. These efforts are complemented by computational fluid dynamic (CFD) studies of viscous supersonic flows as well as by studies of turbulent combustion modeling in the subsonic region of the combustor. Additional studies include the investigation of the stability characteristics of nozzle wall boundary in an attempt to identify methods for controlling transition to turbulence in low Reynolds number nozzles for the purpose of controlling heat transfer and

performance losses. These CFD analyses are also being used for design trade-off studies between radiative and regenerative cooling.

Materials Compatibility Concentration: Our emphasis in materials is directed toward assessing the compatibility of new and existing materials with the harsh environments encountered in propulsion systems. Current emphasis is focused on hydrogen management considerations in conventional and composite materials through the use of multi-layered laminates to control diffusion at high temperatures and pressures and to provide increased materials durability in hydrogen environments. The approaches being considered include the deposition of thin films to provide surface protection and the experimental and theoretical evaluation of the manner in which these films perform in a combustor environment.

Turbomachinery Concentration: The field of turbomachinery represents the newest concentration area of the Center. Our emphasis in this area is on cryogenic bearings, seals and rotor dynamics. Current projects include the development and implementation of an analytical model for predicting the behavior of foil bearings in a cryogenic environment, and a combined experimental/analytical study of magnetic bearings. The foil bearing analysis is concerned with developing and implementing a model for the dynamics and mechanics of the coupled fluid-foil system. The magnetic bearing project includes the experimental set-up of a magnetic bearing facility and emphasizes the design and implementation of advanced closed loop algorithms for controlling the bearing. An advanced non-linear algorithm is being developed to control the dynamics of the magnetic bearing under simulated rocket engine turbopump conditions. An additional turbomachinery project is planned in the area of the hydrodynamic design of cryogenic pumps for liquid rocket engines.

Advanced Propulsion Concepts: The fifth and final research concentration in the Center is on advanced space propulsion concepts and includes research in antiproton annihilation propulsion, microwave propulsion and advanced electric propulsion concepts. The antiproton work includes studies of the feasibility of using antiproton-induced fission fragments to ignite DT pellets for eventual propulsion by inertial confinement fusion. Efforts in microwave and advanced electric propulsion include a vacuum facility for simulated altitude testing and a microwave-plasma facility for propulsion research.

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CRYOGENIC COMBUSTION LABORATORY

CRYOGENIC COMBUSTION LABORATORY

R. J. Santoro and K. K. Kuo
Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

The objective of the current effort is to establish a major experimental laboratory within the NASA Propulsion Engineering Research Center for studying fundamental processes such as mixing and combustion under liquid rocket engine conditions. The capability of this laboratory will include operation using a variety of fuel and oxidizer systems including liquid oxygen and liquid hydrocarbons. In addition to providing the proper facilities for supplying and controlling these fuels and oxidizers, a specific effort is being made to provide a state-of-the-art diagnostic capability for combustion measurements. In particular, optical and laser-based techniques are being emphasized for measurements of species, velocity, temperature, and spray characteristics.

The laboratory is to provide the necessary experimental capabilities for studies conducted within the Center in the area of combustion instability, liquid jet breakup, spray and droplet combustion, supercritical combustion phenomena, materials studies and nozzle flow characterization. The flow and test stand designs emphasize an approach which provides a flexible environment to accommodate a wide range of experiments while providing for convenient future expansion as new research directions emerge.

The Cryogenic Combustion Laboratory is intended to impact space propulsion engineering in a two-fold manner. Experiments conducted in the laboratory are aimed at providing benchmark data on a variety of important liquid rocket combustion processes. A clear need for the development of such data bases for model validation has been recommended in several recent JANNAF workshops. Sub-scale studies utilizing laser-based diagnostic approaches offer a great potential for obtaining new measurements under realistic operating conditions. It is precisely these types of studies which the present laboratory effort is aimed at emphasizing.

The second impact of the current program involves the training of graduate students who participate in the research projects conducted within the laboratory. These students will be exposed to studies involving fuels and oxidizers similar to present operational systems, investigated under conditions applicable to realistic operation. The utilization of sophisticated diagnostic approaches for measurements conducted under combustion conditions will also provide a valuable training experience for the students who will assume positions in leading industrial and government laboratories. Thus, the present laboratory development effort addresses two of the major needs of space propulsion engineering: new fundamental studies of liquid rocket engine phenomena and production of well-trained engineers to engage in future space propulsion activities.

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II. Current Status and Results

During the past year, significant progress has been made in the establishment of the Cryogenic Combustion Laboratory, which had its first test firing using the gaseous hydrogen and oxygen system. This test firing was achieved on schedule and marks one of the major milestones in this project. Future work will proceed on a dual track basis pursuing new experimental results while concurrently adding to the capabilities of the laboratory. Specifically, liquid oxygen and hydrocarbon capabilities are planned to be added in the near future.

The general arrangement and capabilities of the Cryogenic Combustion Laboratory will be reviewed briefly in the remainder of this section. The Cryogenic Combustion Laboratory, space for which was made available to the NASA Propulsion Engineering Research Center in March, 1989, occupies a three-room complex which includes a reinforced concrete test cell (see Figure 1). The test cell is complemented by two adjacent rooms which serve as the instrumentation and control rooms. The control room contains the equipment necessary to operate the test facility. This room is isolated from the test cell during all test run sequences and is equipped with a hydrogen detection system as an added level of safety. The entire system has been designed to operate remotely using pneumatic and/or motorized actuated systems. The design of the fuel and oxidizer supply system closely follows that of an existing facility (Cell 21) located at the NASA Lewis Research Center (LeRC). Personnel at NASA LeRC kindly provided facility drawings and parts lists which greatly aided in the laboratory's development. Additional advice and guidance in the planning of the laboratory has been provided by Marshall Space Flight Center, Rocketdyne, Air Products, the Astronautics Laboratory (Air Force Systems Command) and Aerojet, and their input is gratefully acknowledged.

The operation of the test facility presently allows for operation with gaseous oxygen and gaseous fuels (hydrogen and methane have been used to date). The maximum operating pressure is approximately 1500 psi, a limit imposed by the present fire valve. The flow system has been designed to provide a maximum flow rate of 0.1 lbs/s of oxygen. This has been decided to be adequate for the present sub-scale studies for which this laboratory is intended. The sequencing of all combustion test firing is controlled by a Modicon sequencer which provides real time fault detection for run monitoring and shutdown. System testing of the entire flow system was achieved using a rocket igniter constructed by the NASA LeRC. This unit has been widely used at the NASA LeRC and represented a known test engine from which to evaluate the operation of gaseous flow system and control equipment. Successful firings of this igniter occurred near the end of December, 1989, and have periodically continued through March, 1990. Both hydrogen/oxygen and methane/oxygen test runs have been conducted. This extensive series of tests has provided a suitable basis for assuring the adequacy of the safety systems, run procedures and general operation of the laboratory. Figure 2 shows a photograph of one of the test firings of the rocket igniter burning methane/oxygen.

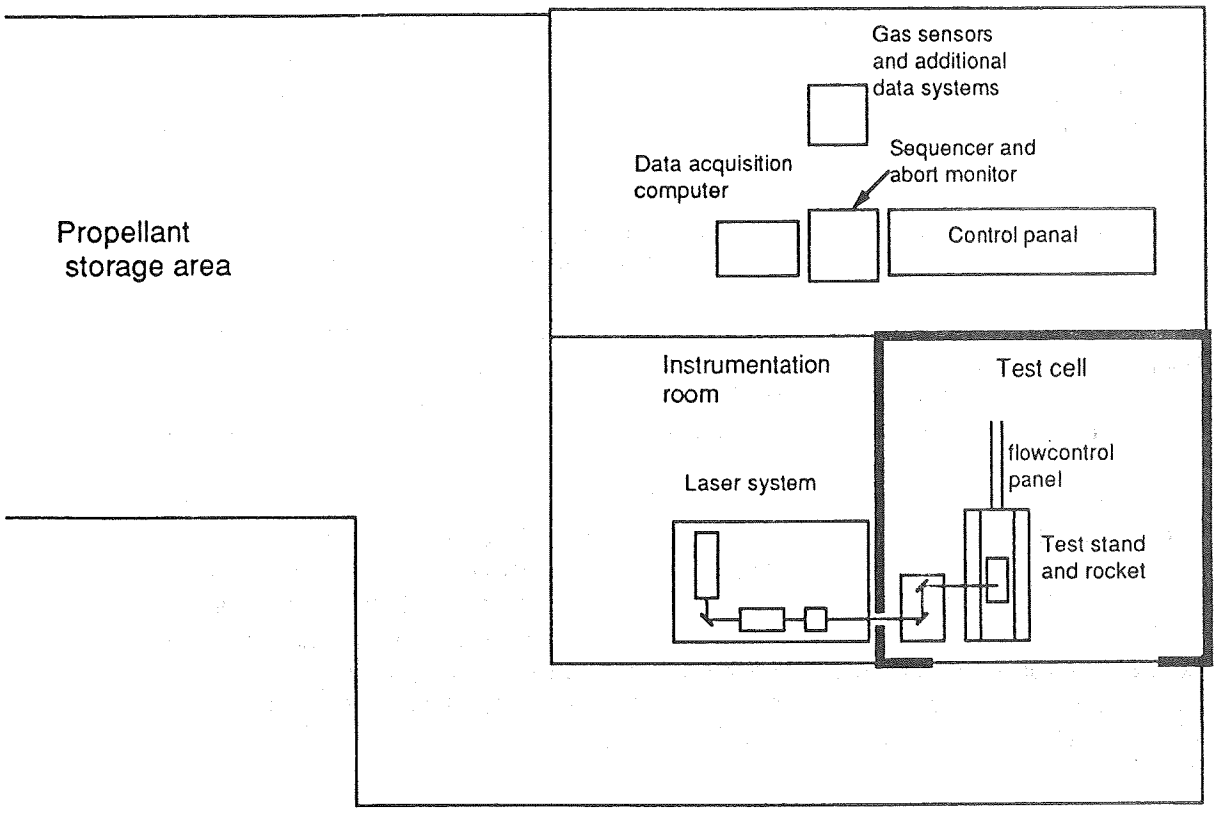


Figure 1. Schematic representation of the Cryogenic Combustion Laboratory Complex: TestCell, Instrumentation Room and Control Room

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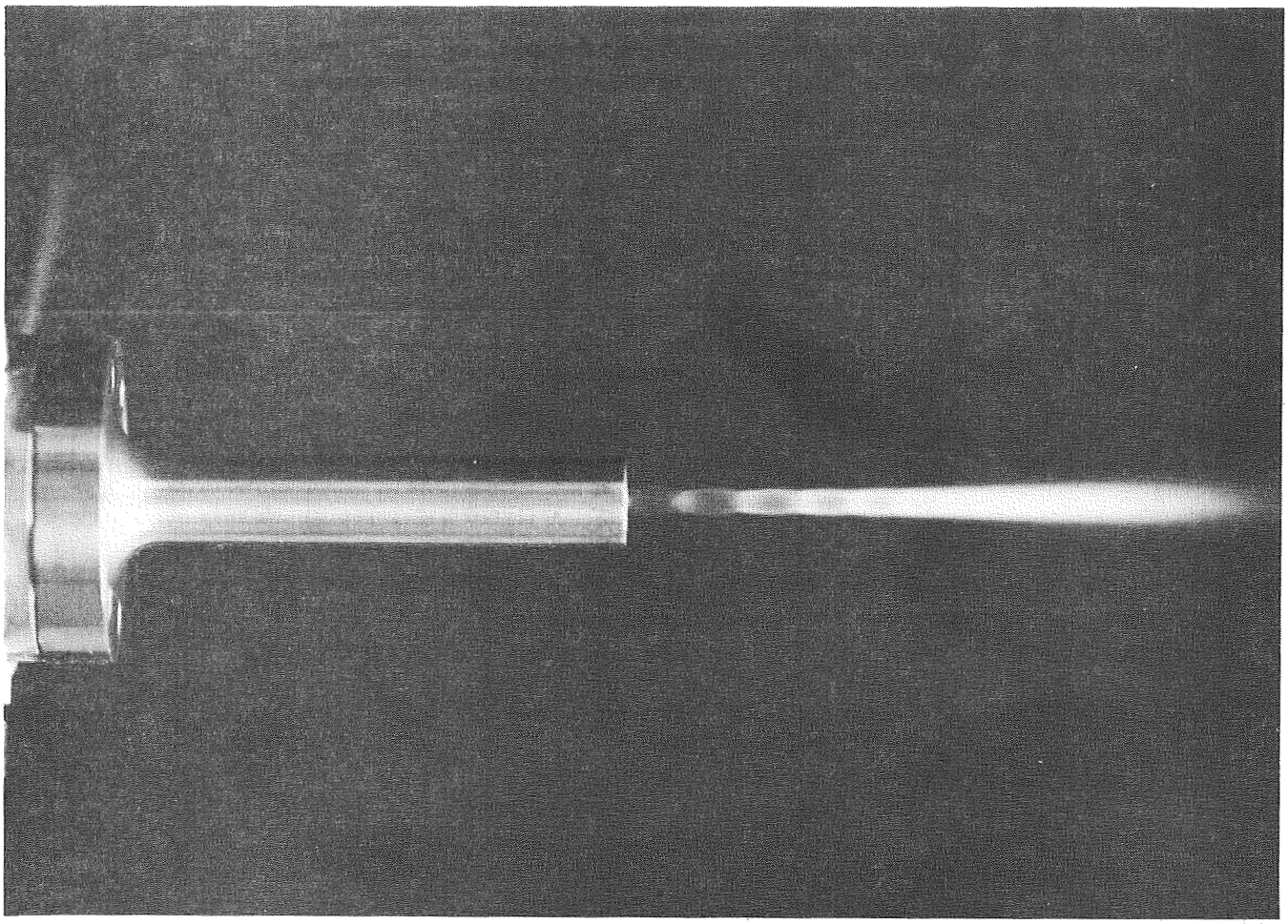


Figure 2: Rocket Test Igniter Firing: Methane oxygen test at an equivalence ratio of 0.56. Oxidizer mass flow rate of 0.01 lbm/sec, fuel mass flow rate of 0.0014 lbm/s. Chamber pressure 85 psia.

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In addition to the progress achieved in the gaseous flow systems for the laboratory, efforts have been initiated to add a liquid oxygen capability. A suitable cryogenic oxygen storage tank has been obtained. This tank was originally associated with Prof. L. Crocco's laboratory at Princeton University and has been kindly made available to the NASA Propulsion Engineering Research Center by Princeton. The initial design plans for the liquid oxygen flow system have been completed and the components are presently being ordered.

Future plans for experiments using the Cryogenic Combustion Laboratory include imaging studies of the combustion processes within the rocket chamber. These studies will initially concentrate on planar (i.e. two-dimensional) imaging of OH radical concentration profiles in a hydrogen/oxygen rocket. Further studies of spray combustion processes are also planned using both planar and point measurement techniques. Efforts to establish the necessary diagnostic capabilities have also progressed substantially. Both a pulsed Nd-Yag and a cw argon ion laser system have been acquired during the last year for use in these studies. Diagnostics for providing two-dimensional imaging of droplets in a combusting environment are presently being developed in a separate laboratory for use with these lasers. Additionally, a phase doppler particle analyzer has recently been added for additional droplet sizing capabilities. A dye laser will also be available shortly to give additional wavelength selection capabilities.

In summary, the first phase of the Cryogenic Combustion Laboratory has been successfully completed. Progress on applying the present test capability to new measurements and studies has been initiated. Planning to complete the second major operational goal, liquid oxygen capability, is proceeding on schedule. Additionally, a solid diagnostic basis is being developed to complement the test capability of the laboratory with state-of-the-art diagnostic techniques.

III. Proposed Work for Coming Year

During the coming year, the proposed work will focus on augmenting the Cryogenic Combustion Laboratory to include liquid oxygen capability. The present design specifies a LOX mass flow rate of 1 lb/s which significantly increases the present capabilities over that of the gaseous oxygen system. The intent of achieving this flow rate is based on a desire to adequately study liquid rocket injector elements. Both uni-element and multi-element (three to five elements) studies should be feasible with the planned flow system. Initial testing of this system will be accomplished using the current hydrogen fuel system, and initial tests are planned for the end of the present calendar year.

As a concurrent effort, the first two of a series of laser diagnostic techniques are presently under development. These techniques involve planar laser imaging of OH and spray visualization and sizing. Both techniques are intended to provide spatially extensive (i.e. two-dimensional fields) under high temporal resolution conditions. The OH diagnostics will utilize the Center's present Nd-Yag laser in conjunction with a recently acquired dye laser and wavelength doubling system. The necessary data acquisition and solid-state camera system are presently being specified. Initial development of the OH visualization system will be accomplished using a flat flame burner before application to a uni-element injector, rocket combustor environment. Presently, several design approaches are being pursued for developing an appropriate optically accessible rocket chamber. Initial measurements of OH in such a rocket combustor are planned for the coming year of the program. These initial rocket studies will utilize the present gaseous hydrogen and oxygen flow system. Use of this system provides the simplest system for achieving the planned measurements, eliminating interference from droplets or soot particles. The intention of the present program is to examine the OH distribution as a qualitative indication of the combustion zone distribution in the combustor. Future work is intended to examine the effects of oscillating pressure fields on that distribution. However, these efforts go beyond the present program for the Cryogenic Combustion Laboratory which emphasizes development of the basic laboratory and diagnostic capability to support the space propulsion research interests of the Center.

In a similar approach, a planar laser imaging approach for spray diagnostics is being developed as part of a separate program on rocket spray combustion phenomena (H.R. Jacobs and R.J. Santoro, "Spray Combustion Under Oscillatory Pressure Conditions".) This technique utilizes the ratio of scattered light obtained at two polarization orientations to provide a measure of the droplet size. Evaluation of this diagnostic approach is presently proceeding in a non-combusting environment. Based on the results of these studies, this approach will be implemented for the study of liquid oxygen sprays in coaxial injector elements.

In summary, the proposed effort for the current year involves both a continued development of the Cryogenic Combustion Laboratory along with the initiation of basic propulsion studies in simple rocket combustors. These developments represent the realization of a laboratory capability needed for future Center research efforts.

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IGNITION AND COMBUSTION OF METALLIZED PROPELLANTS

IGNITION AND COMBUSTION OF METALLIZED PROPELLANTS

Stephen R. Turns
Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

The overall objective of our research is the development of a fundamental understanding of the ignition and combustion of aluminum-based slurry (or gel) propellant droplets using a combination of experiment and analysis. Specific objectives are the following:

1. The development and application of a burner/spray rig and single particle optical diagnostics to study the detailed ignition and combustion behavior of small (10-75 μm) droplets, typical of those encountered in practical applications.

2. Understanding the role of surfactants and gellants (or other additives) in promoting or inhibiting secondary atomization of propellant droplets.

3. The extension of previously developed analytical models and the development of new models to address the phenomena associated with microexplosions (secondary atomization).

Slurry or gelled propellants in which a solid constituent is suspended in a combustible liquid have been of interest for propulsion applications for several decades. Depending upon the application, these propellants are advantageous because of either their energy content per unit mass or per unit volume. Many solid chemical elements are attractive for formulating slurry or gelled propellants. For example, carbon, aluminum, and boron all offer substantial increases in volumetric heat of combustion compared to hydrocarbon fuels, while on a gravimetric basis, only boron provides an increased heating value [1].

For rocket applications, specific impulse and mass density control the payload capability for a fixed vehicle configuration. Thus, high mass density propellant systems, such as Al/RP-1/0_2, become attractive to consider as alternatives to more conventional propellant systems. Zurawski and Green [2] show that an RP-1/60% Al fuel has the potential of delivering 17% more payload than a pure RP-1 fuel for an upper-stage vehicle mission.

Several key technical issues impact the use of Al/RP-1 slurries or gels in propulsion systems. Among these are rheological properties and ignition and combustion characteristics. In the area of rheology, questions concerning the proper formulation to provide a stable easily pumpable slurry with acceptable spray characteristics remain unanswered. With regard to ignition and combustion, the issue of concern is the relatively long residence times associated with these processes and two-phase flow losses. These issues are the focus of our research. We have found in previous studies of Al/JP-10 slurries that the additives used to produce desirable

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rheological properties impact on the ignition characteristics [3]. For example, the addition of surfactants to a slurry blend tend to promote the break up of burning slurry droplets into a very large number of smaller droplets. The aluminum in these small drops readily ignites and burns rapidly. In addition, the particle sizes of the aluminum oxide products are smaller, thus decreasing two-phase flow losses. Achieving the research objectives will provide detailed knowledge of the effect of blending agents on ignition and combustion of slurry fuels. This information will help lead to formulation of propellants which provide an optimum combination, or an intelligent trade-off, of rheological and combustion characteristics. Moreover, understanding and being able to predict ignition times for small droplets will provide guidelines for feasibility and developmental studies of metallized propellant engine systems prior to the construction of costly demonstration hardware.

II. Current Status and Results

Our efforts to date have focused primarily on the design, fabrication, set-up, and calibration of the various experimental systems required to study the ignition and combustion of 10-100 μm diameter slurry droplets. The major systems are the burner/spray rig and the optical diagnostics. Schematics of these systems are shown in Figs. 1 and 2, respectively. The design objectives for the burner/spray rig were to provide a laminar, homogeneous post-flame region for slurry ignition; to allow flashback-free operation over a range of stoichiometries while operating with oxidizers ranging from air to 100% oxygen; to isolate the post-flame gases from the ambient atmosphere; to produce a spray of aluminum slurry droplets varying in size from approximately 10 to 100 μm in diameter and introduce a portion of this spray into the post flame region; and to provide for fixed mounting of laser optics through the use of a traversing burner support. A burner meeting these requirements was designed, fabricated, and installed in the laboratory. The system is operational, and its detailed performance is currently being evaluated. The burner head is based on a design discussed in Ref. [4]. Gaseous fuel enters the base of the burner and passes through a dispersion ring that evenly distributes the gas around the perimeter of a fuel chamber. From here, the fuel passes through 72 stainless steel tubes and exits the burner at the top surface of a honeycomb matrix. The oxidizing gas enters the middle section of the burner and passes through another dispersion ring above the manifold plate. This gas then flows up around the 72 fuel tubes and through the open cells of the honeycomb matrix surrounding these tubes. This configuration results in a small, laminar, diffusion flame at the exit of each fuel tube. These flames merge rapidly providing an excellent approximation of a pre-mixed, laminar, flat flame. Slurry droplets generated in the spray chamber at the bottom of the burner pass through a tube located along the centerline of the burner and exit into the hot product gases. The spray nozzle is of the gas atomizing type and was selected based on work with coal/water slurries [5]. Initial testing with water showed that the system delivers droplets of the desired size range into the hot product stream.

The principal diagnostic being used to study droplet ignition, burning, and secondary atomization is a laser-based sizing and velocity measurement system. By measuring the evolution of the drop-size distribution with distance downstream of the burner face, we will be able to ascertain whether the droplets burn without disruption

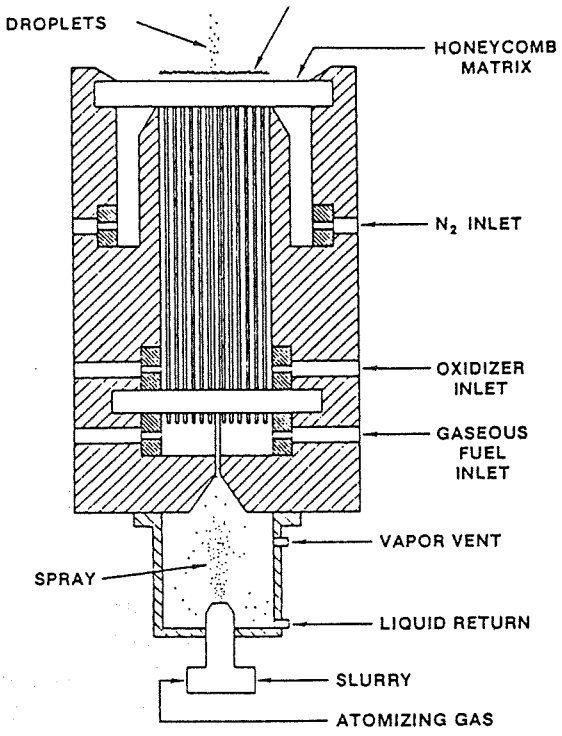


FIG. 1. Burner and spray rig.

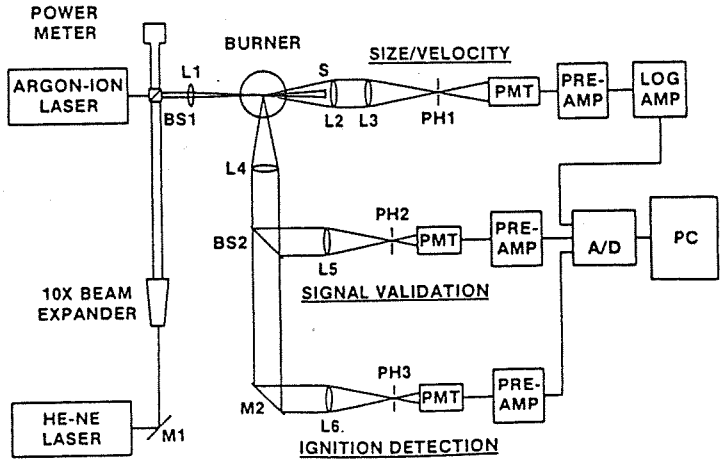


FIG. 2. Optical system for particle sizing and velocity measurements.

or if they disrupt. The laser light scattering system designed and built to accomplish these measurements is shown in Fig. 2. In principle, the two-color laser light scattering technique [6] operates as follows: droplets pass through the 350 μm diameter waist of a focused Ar-ion laser beam. Light scattered by the droplets in the forward direction is related to particle size, with larger particles producing a larger signal. Because of the Gaussian distribution of light intensity in the laser beam, only data from droplets that pass through the beam center should be accepted. This eliminates the ambiguity associated with the light scattered by droplets passing through the edge of the beam. For example, a large droplet at the edge of the beam would scatter the same amount of light as a small droplet passing through the center. To accomplish this discrimination, a second beam from a He-Ne laser is aligned concentrically with the Ar-ion beam, but focused to a much smaller waist of approximately 80 μm. Comparison of the time-dependent signals associated with scattering of the He-Ne and Ar-ion beams determines whether the droplet passed through the central region of the Ar-ion beam.

The determination of the particle size is accomplished by a calibration of the scattered intensity associated with light diffracted through pinholes of known size and by performing measurements in a stream of monodisperse water droplets. Droplets of known size are generated with a Berglund-Liu droplet generator. Results of a typical droplet size calibration are shown in Fig. 3. In addition to obtaining information on droplet size, knowing the beam diameter readily permits the calculation of droplet velocity from the droplet time-of-flight through the beam. Figure 4 illustrates a typical voltage-versus-time trace for a 90 μm water droplet passing through the probe volume. The peak voltage of approximately 5.5 volts from the upper trace provides the size information (c.f. calibration curve in Fig. 3.). The droplet velocity computed from the width of the trace is 1.3 m/s. The signal validation pulse associated with the light scattered from the concentric red beam is the smaller signal centered below the primary scattering signal.

III. Proposed Work for Coming Year

During the next 12-month period, the following tasks are planned:

Task 1. Completion of software development for data acquisition and signal processing.

Task 2. Integrated operation of the complete system (burner, spray rig, particle sizing and velocimetry systems, data acquisition, and data processing) using pure liquids and aluminum slurry fuels.

Task 3. Perform screening study to establish which slurry formulation should be used as a baseline.

Task 4. Conduct parametric study using baseline slurry.

Task 5. Begin modeling effort.

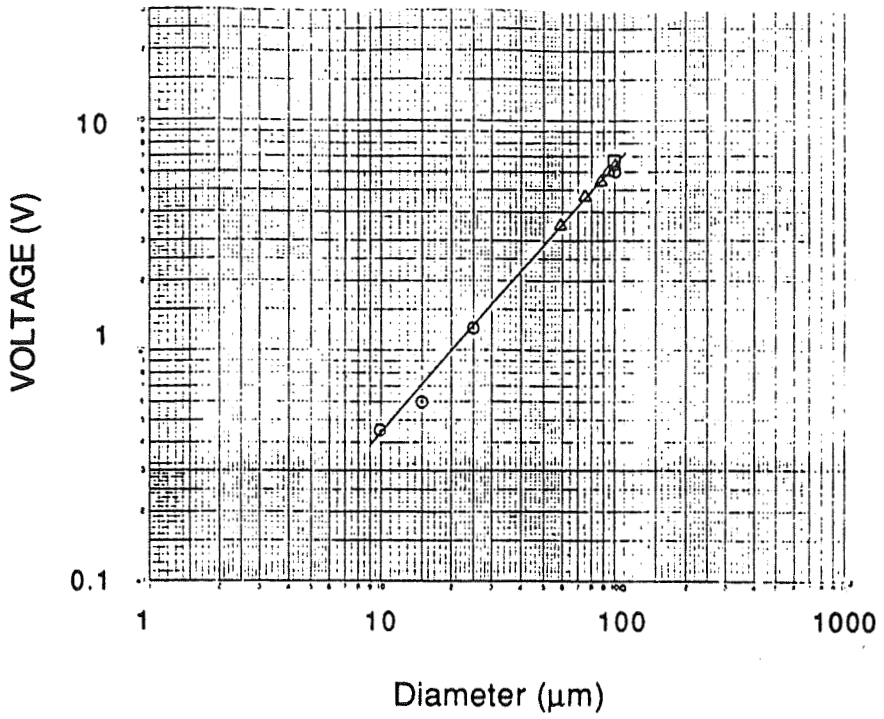


FIG. 3. Particle sizing calibration.

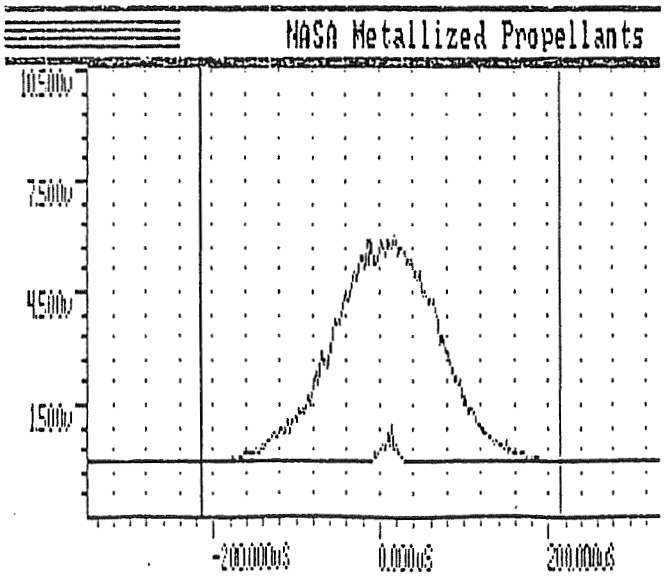


FIG. 4. Voltage-vs.-time trace for water droplet passing through probe volume.

Task 1. At present, our data acquisition capability is limited to obtaining voltage-versus-time records for the scattering signals associated with the transit of a single particle through the focal volume. Software for a main menu control has been written which provides set-up of the acquisition parameters for all A/D boards. The following describes the software that will be completed in the early portion of the upcoming year. The data collection software performs the signal validation and records the pulse height and width of the Ar-ion signal and whether or not the particle was ignited. Signal validation involves checking for the presence of multiple particles in the probe volume, peak saturation, pulse shape, and whether the Ar-ion pulse is completely contained within the data sample taken. The program will only accept complete single particle signals for analysis, rejecting all other data. The peak and the leading and trailing edges of the Ar-ion signal also are located at this time. The Ar-ion signal pulse width is then calculated, and the pertinent information is passed to storage. This extraction of the critical information from the signal drastically reduces the disk space required to define a single particle, making possible the storage of a large number of samples. The data analysis portion of the software reads particle data from a specified file and performs the necessary sizing and velocity calculations based on drop-size calibrations of the laser system. After this is done, the results are written to a second data file for further statistical and graphical analysis.

Task 2. This task is the final shakedown of the complete experimental system. Initial experiments will be conducted with pure liquids to simplify fuel handling. The burner operation limits will be investigated to determine maximum and minimum gas velocities that can be achieved with different mixture fractions and oxygen content of the oxidizer stream.

Task 3. At present, three Al/hydrocarbon gelled fuels are available: a 55 wt. % Al/RP-1 fuel formulated by Sun Advanced Research and Marketing; a 53.8 wt. % Al/RP-1 blend formulated by TRW Space & Technology Group; and a 55 wt. % Al/RP-1 gelled fuel from Aerojet TechSystems. Different blending agents were used in each of these formulations, so each fuel is likely to exhibit different secondary atomization characteristics. Tests will be conducted on each blend to ascertain which fuel provides the best secondary atomization characteristics for a limited set of test conditions. The fuel selected will be used in the following task.

Task 4. Using the baseline fuel, an extensive set of parametric experiments will be carried out. These experiments will explore the effects of the hot gas environment on secondary atomization. The parameters to be investigated include: gas temperature, oxygen concentration, and residence time. A second set of experiments will be planned to investigate the effects of the minor constituents in the fuel blends on secondary atomization.

Task 5. In support of the experimental efforts in Task 4, data will be analyzed using codes which predict droplet ignition and combustion times. Existing codes are presently being modified for this task. A physical model for crust formation on slurry droplets during the RP-1 burnout phase will be formulated, and the appropriate conservation laws applied.

IV. References

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PRESENTATION 2.2.4

N91-28225

LIQUID-PROPELLANT DROPLET VAPORIZATION AND COMBUSTION IN HIGH PRESSURE ENVIRONMENTS

LIQUID-PROPELLANT DROPLET VAPORIZATION AND COMBUSTION IN HIGH PRESSURE ENVIRONMENTS

Vigor Yang
Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

Many practical liquid-propellant rocket propulsion systems involve droplet vaporization and spray combustion in high-pressure environments. Liquid propellants are usually delivered to combustion chambers as a spray of droplets, which then undergo a sequence of vaporization, ignition, and combustion processes at pressure levels well above the thermodynamic critical points of the liquids. Under these conditions, droplets initially injected at subcritical temperatures may heat up and experience a thermodynamic state transition into the supercritical regime during their lifetimes. Consequently, the sharp distinction between gas and liquid disappears, and the entire system exhibits many distinct characteristics which conventional droplet theories developed for low-pressure cases can not deal with.

Attempts to study droplet vaporization and combustion in the supercritical regime have been made for three decades. While these studies have provided significant information to understanding the physics and chemistry of droplet vaporization and combustion at high pressures, a number of fundamental problems remain unresolved. Much of the previous work employed certain assumptions and empirical correlations which were extrapolated from low-pressure cases and involved a considerable number of uncertainties. In order to correct the deficiencies of existing models for high-pressure droplet vaporization and combustion, a fundamental investigation into this matter is essential. The specific objectives of this research are:

1. to acquire basic understanding of physical and chemical mechanisms involved in the vaporization and combustion of isolated liquid-propellant droplets in both stagnant and forced-convective environments;

2. to establish droplet vaporization and combustion correlations for the study of liquid-propellant spray combustion and two-phase flowfields in rocket motors; and

3. to investigate the dynamic responses of multicomponent droplet vaporization and combustion to ambient flow oscillations.

II. Current Status and Results

During this report period, emphasis has been placed in three areas: (1) assessment of constitutive relations and property evaluation techniques for multicomponent mixtures at high pressures; (2) analysis of droplet vaporization at near

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and supercritical conditions; and (3) investigation of droplet combustion at near and supercritical conditions. The major status of our work is given below.

1. Assessment of Constitutive Relations and Property Evaluation Techniques for Multicomponent Mixtures at High Pressures

One of the fundamental difficulties in the study of supercritical droplet vaporization and combustion is the lack of reliable data on thermodynamic and transport properties of each constituent species as well as of the mixture at high pressures. The problem becomes even more severe at near-critical conditions (within a few percents of the critical temperature). The thermophysical properties usually exhibit anomalous behavior, and become very sensitive to both temperature and pressure due to a transition of molecular ordering or to small-scale circulation effects resulting from the migration of clusters of molecules . When generalized correlations are used, these property irregularities are smoothed out and may lead to inaccurate results. During the last year, efforts have been made continuously to assess (and occasionally establish) various property evaluation techniques and constitutive relations for multicomponent mixtures at high pressures. These relations include equations of state and mixing/combining rules with high-pressure corrections. In addition, a generalized methodology and an efficient numerical algorithm were developed to treat thermodynamic vapor-liquid phase equilibria for liquid propellants at high pressures. Specific outputs from the phase-equilibrium analysis include:

- latent heat of vaporization
- solubility of ambient gas in the liquid phase
- liquid and gas-phase species concentrations at the droplet surface

2. Analysis of Droplet Vaporization at Near and Supercritical Conditions

The specific objective of this work is to analyze, from first principles, the detailed physical processes involved in multicomponent liquid-propellant droplet vaporization at high pressures. The formulation is based on the full time-dependent conservation equations, and accommodates a thorough treatment of property variations and vapor-liquid phase equilibrium. Because of its completeness, the model enables a systematic examination of the droplet vaporization characteristics in a high-pressure environment. In particular, the effects of ambient gas solubility, property variation, transient diffusion, and multicomponent transport on the droplet behavior are investigated in depth.

3. Investigation of Droplet Combustion at Near and Supercritical Conditions

The analysis of droplet combustion extends the model for droplet vaporization and accommodates finite-rate chemical kinetics. Consequently, various combustion related problems (such as ignition, flame development, extinction, etc.) can be treated in detail. Specific results of this task include:

1. thermodynamic conditions for reaching supercritical combustion;

2. complete time history of the flowfield and interface transfer rates for evaporating or burning droplets at various ambient conditions; and

3. ignition criteria for a variety of liquid-propellant and ambient-gas situations.

The influences of volatility and miscibility of the liquid constituents on droplet vaporization and combustion characteristics are also addressed.

Calculations were first carried out to study the combustion process of a single droplet in a stagnant air environment, due to the availability of experimental data for model validation. At t = 0, a pure n-pentane droplet with a diameter of 100 μm is exposed suddenly to air. The initial droplet and air temperature are 300 and 1200 K, respectively. Figure 1 shows the distributions of the gas-phase temperature at various times. The air pressure is 50 atm. Initially, as a result of heat feedback to the droplet, the liquid fuel starts to evaporate. The resultant vapor then ignites with the ambient air, causing a small hump in the temperature profile at t = 1 msec. The rapid increase in the gas-phase temperature in the subsequent stage suggests the onset of flame development. Figure 2 presents the corresponding temperature distributions in the liquid phase. The penetration of thermal wave in the droplet is clearly observed.

Figure 3 presents the histories of the square of droplet diameter at various pressures. The droplet surface regression rate increases with pressure mainly as a result of reduced heat of vaporization at high pressures. In the initial period, the droplet diameter decreases slowly with only vaporization involved in the regression process. The regression rate then increases rapidly immediately after the onset of flame development in the gas phase, due to the enhanced heat transfer to the liquid phase.

The analysis was compared with the conventional low-pressure model in which Raoult's law is employed to determine the species concentrations at the droplet surface before the occurrence of the critical state. In addition, the ambient gas solubility is ignored and the latent heat of vaporization is taken to be that of the saturated pure liquid. Figure 4 presents the time variations of the square of the droplet diameter at p = 50 atm in accordance with both the high- and low-pressure models. Because of its inability to predict the correct droplet surface condition, the low-pressure model overpredicts the surface temperature, thereby leading to excessive volumetric dilatation. However, this effect is offset by the underpredicted latent heat of vaporization in the later stage of the droplet lifetime and consequently causes a faster regression rate. In regard to the mass evaporation rate, the discrepancy between these two models appears to be even greater due to the lower liquid density predicted by the low-pressure model.

VI. Proposed Work for Coming Year

Future work on high-pressure droplet vaporization and combustion will consist of the following tasks.

1. Vaporization and Combustion of Cryogenic Propellant Droplets

The specific objective of this task is to enhance our current understanding of cryogenic propellant droplet combustion at supercritical conditions. The work represents an outgrowth of the analysis developed in the last year and incorporates several unique characteristics of cryogenic propellants into the model. Special attention will be focused on the evaluation of thermophysical properties and establishment of constitutive relationships. Representative propellant combinations (such as LOX/RP-1 and LOX/LH_2) will be treated to simulate actual motor conditions.

2. Dynamic Responses of Droplet Vaporization and Combustion to Ambient Flow Oscillations

The purpose of this work is to identify some of the combustion instability driving mechanisms associated with supercritical droplet vaporization and combustion. The analysis will begin with essentially the same as that for a steady environment, but with appropriate modifications of the outer boundary conditions for the gas phase, various pressure-coupled combustion response functions can be obtained. Cases of both pulsed and periodic oscillations will be examined carefully. Results will be used to establish useful correlations for existing analyses of linear combustion instabilities in rocket motors.

3. Droplet Vaporization and Combustion in a Forced Convective Environment

The primary objective of this task is to investigate the effects of forced convection on droplet vaporization and combustion processes. The work will extend the analysis developed in the last year and solve the multi-dimensional flowfield surrounding the droplet. Results can be used to assess and/or establish useful correlations for heat, mass, and momentum transfer rates under forced convective situations. Special attention will be paid to the applications to the study of liquid-propellant spray combustion and two-phase flowfields in rocket motors.

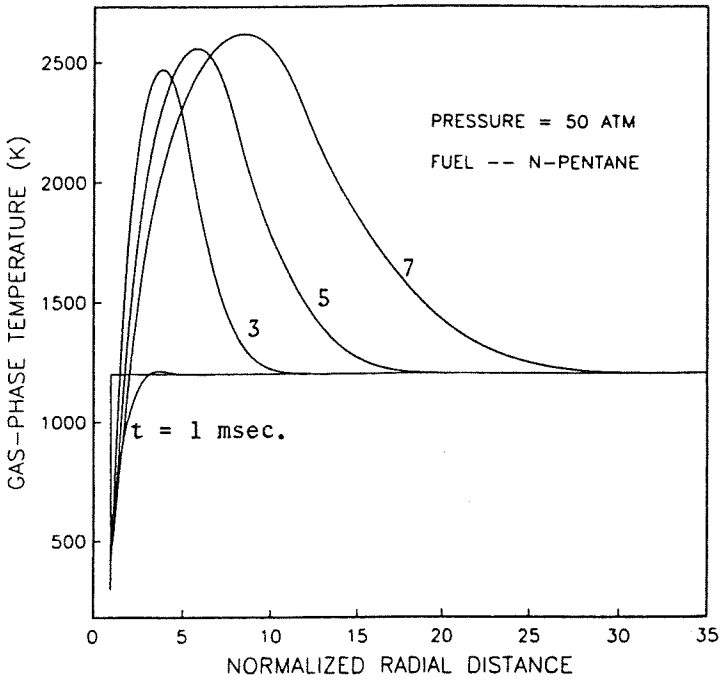


Fig. 1. Distributions of Gas-Phase Temperature at Various Times.

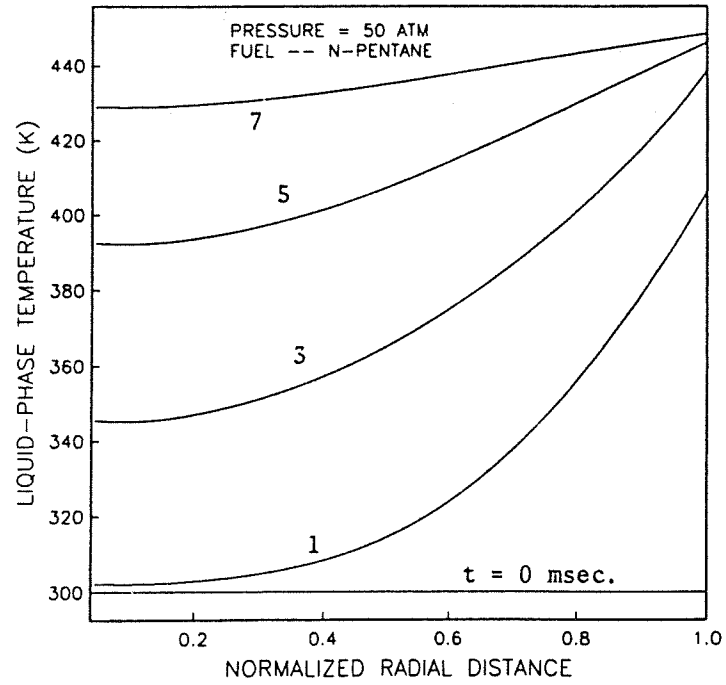


Fig.2. Distributions of Liquid-Phase Temperature at Various Times.

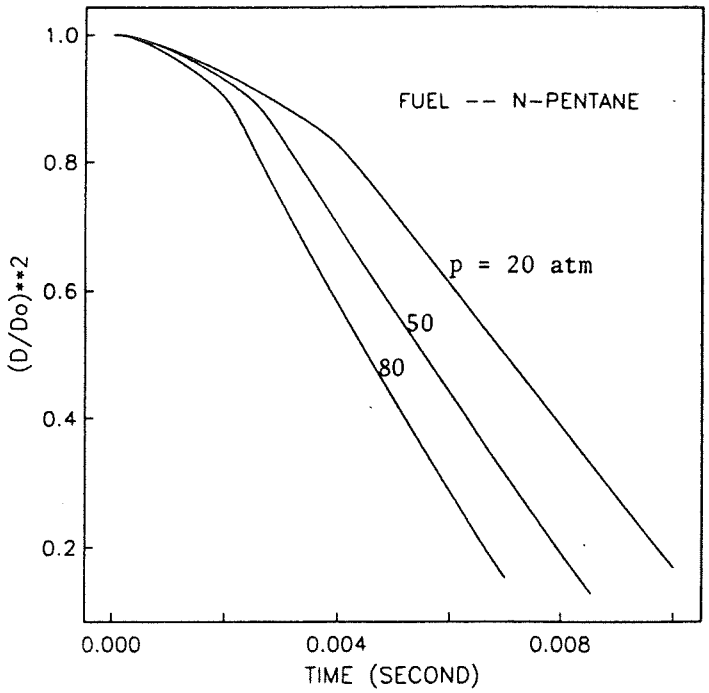


Fig. 3. Time Histories of the Square of Droplet Diameter at Various Pressures.

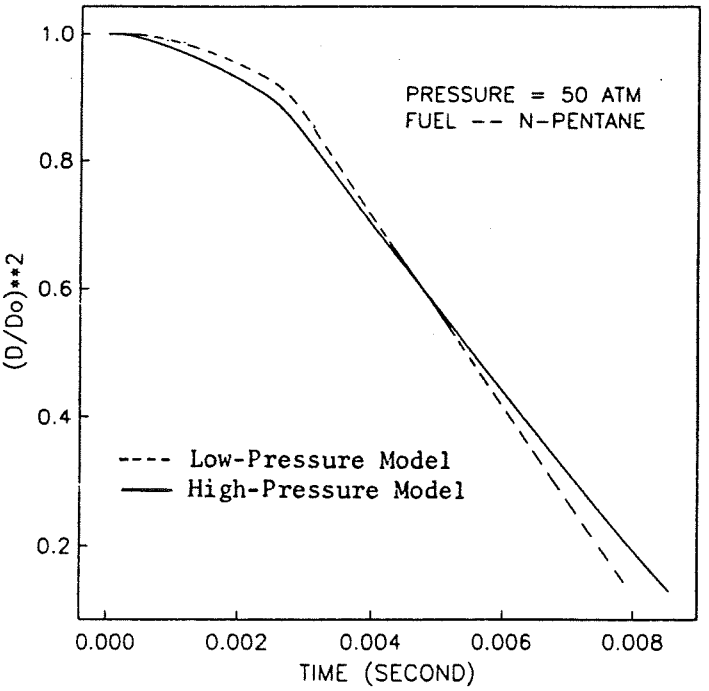


Fig. 4. Time Histories of the Square of Droplet Diameter at p = 50 atm.

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PRESENTATION 2.2.5

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SPRAY COMBUSTION UNDER OSCILLATORY PRESSURE CONDITIONS

SPRAY COMBUSTION UNDER OSCILLATORY PRESSURE CONDITIONS

H. R. Jacobs and R. J. Santoro
Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

The performance and stability of liquid rocket engines is often argued to be significantly impacted by atomization and droplet vaporization processes. In particular, combustion instability phenomena may result from the interactions between the oscillating pressure field present in the rocket combustor and the fuel and oxidizer injection process. Few studies have been conducted to examine the effects of oscillating pressure fields on spray formation and its evolution under rocket engine conditions. The present study is intended to address the need for such studies. In particular, two potentially important phenomena are addressed in the present effort. The first involves the enhancement of the atomization process for a liquid jet subjected to an oscillating pressure field of known frequency and amplitude. The objective of this part of the study is to examine the coupling between the pressure field and/or the resulting periodically perturbed velocity field on the breakup of the liquid jet. In particular, transverse mode oscillations are of interest since such modes are considered of primary importance in combustion instability phenomena.

The second aspect of the project involves the effects of an oscillating pressure field on droplet coagulation and secondary atomization. The objective of this study is to examine the conditions under which phenomena following the atomization process are affected by perturbations to the pressure or velocity fields. Both coagulation and secondary atomization affect droplet vaporization processes and consequently can represent a coupling mechanism between the pressure field and the energy release process in rocket combustors. It is precisely this coupling which drives combustion instability phenomena. Consequently, the present effort is intended to provide the fundamental insights needed to evaluate these processes as important mechanisms in liquid rocket instability phenomena.

Due to the challenging measurement environment presented by these studies, a complementary diagnostic development effort has proceeded in parallel with the above studies. These diagnostic approaches are emphasizing real-time measurements of droplet size, size distribution and spatial location. Both novel, as well as state-of-the-art, techniques are presently being addressed and validated in the present experiments.

The major impact of the present study on propulsion engineering involves the potential for gaining a fundamental understanding of the role of pressure variations on atomization and spray phenomena. Through an understanding of the proposed mechanisms for combustion instability generation, the basic understanding of the underlying physics required for development of appropriate submodels can be achieved. Present rocket combustion modelling efforts suffer from the lack of appropriate

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submodels for describing the interactions between spray formation and droplet processes with oscillating pressure and velocity fields. The present research program is aimed directly at resolving some of the critical elements in that process.

II. Current Status and Results

A. Jet Breakup Under Acoustic Pressure Oscillations

A sequence of experiments have been conducted to study the effects of acoustic oscillations on the breakup of liquid jets and the trajectories of the resulting droplets. Both mono-injector (water injection) and co-axial nozzles (water core; nitrogen annulus) were investigated towards this end. Experimental techniques used for this investigation involved high speed cinematography (Spin Physics camera), laser light scattering/polarization ratio techniques and simple flash photography. A new innovative technique that will be used in the near future is phase-doppler anemometry for droplet sizing. Experimental results for the mono-injector nozzles show that high frequency acoustic oscillations (1-4 kHz) play a dramatic role in the breakup of liquid jets at certain preferential modes that are characteristic of the injection chamber. These results are of potential importance for impinging-element type injectors for obvious reasons. Initial experimental results for the co-axial nozzle have shown no dramatic changes in liquid atomization characteristics. However, measurement capabilities have been limited in these initial experiments and more exhaustive consideration is to be given to effects on the droplet field in the near future. Preliminary numerical calculations indicate that the trajectories of small droplets (less than approximately 100 μm) are significantly affected both by acoustic oscillations and the phase angle of the oscillations at the moment of atomization. Numerical calculations also show that these small droplets are subjected to extremely high g-forces, which suggests that acoustic oscillations can trigger secondary atomization phenomenon.

Experimental Setup

A schematic of the cylindrical chamber used for our experiments is shown in Fig. 1. The 6 inch diameter, 24 inch steel chamber has two 120-Watt Altec-Lansing speakers attached at the ends of the speaker arms. These speakers are used to drive the acoustic modes in the chamber. The speakers can be driven with any desired phase separation and microphone measurements of the pressure field within the chamber show that peak to peak oscillations in excess of 4 psi can be maintained. Two windows, centered 6 inches from the top of the chamber provide visual access. Optionally, one of the speakers can be replaced with a window to provide additional visual access. A moveable injector as shown in Fig. 1 is used to position the injector face near the window for added visual access. Four circumferential microphone ports, at 90° intervals are located 10 inches from the top of the chamber. Microphones employed at these ports are used to study the acoustic characteristics of the chamber.

The acoustic modes of the chamber were first identified both analytically and experimentally. The standing wave frequencies were calculated by solving the 3-D wave equation by separation of variables. For the experimental identification of the standing wave frequencies, a white noise generator was used to drive the speakers and the resulting microphone responses at the various ports were analyzed. Good agreement between the calculations and experimental results were found for the simple longitudinal and tangential modes. The higher combined modes of the chamber were, however, more difficult to identify and is a reflection of the rich frequency characteristics of the chamber.

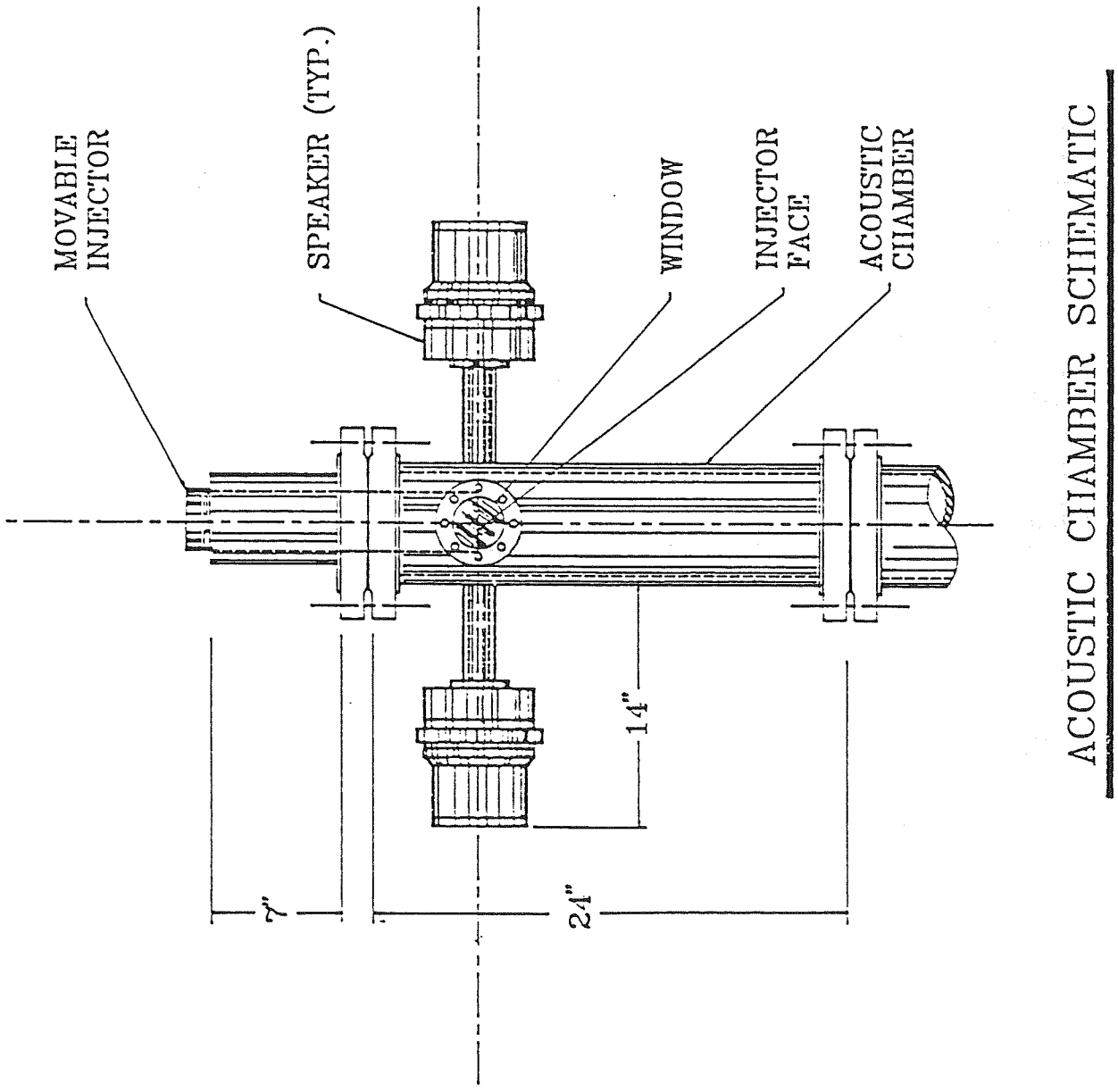


Fig. 1. A schematic of the injection chamber.

Jet Breakup Results

Acoustic oscillations were visually and photographically observed to dramatically breakup the liquid jet emanating from mono-injector nozzles (0.0625 inch and 0.1 inch diameter nozzles with a length to diameter ratio in excess of 10) in two distinct fashions. The Reynolds numbers of the jets ranged from 2000 to 9000 and the corresponding Weber numbers based on the gas density ranged from 0.03 to 0.5, which places these jets in the Rayleigh breakup region. The first type of breakup occurred at 1140 Hz which corresponds to the first tangential mode. The jet was observed to breakup into a spray with droplet diameters of the same order of magnitude as the nozzle diameter. A photograph of this tangential-mode breakup is shown in Fig. 2. The corresponding pressure amplitude pattern at one phase angle measured by traversing two microphones within the chamber is shown in the inset graph. Note that the centerline of the chamber that bifurcates the speaker axis also delineates the positive and negative pressure amplitudes. This type of pressure pattern indicates that the mode is 1-T mode. It is postulated that the 1-T mode frequency couples preferentially to the breakup frequencies of the jet. The second type of breakup is shown in Fig. 3 at a frequency of 1560 Hz. For this type of breakup, the jet first acquires the shape of a two-dimensional fan (perpendicular to the speaker axis) which is reminiscent of the fan-structure observed with impinging nozzles. Droplets sheared from the bottom of the fan are visually at least more than an order of magnitude smaller than the diameter of the nozzle. Acoustic measurements of the mode in question show tangential mode-like characteristics as can be seen from the inset graph in Fig. 3. However, the sharp pressure gradient that exists along the center vertical plane of the chamber that seems to cause the jet to 'fan' out is not characteristic of a pure tangential mode. This mode could be the 3-Longitudinal/1-Tangential (calculated to be 1580 Hz) mode. Note that the pressure amplitude measurements were made 10 inches from the top of the chamber which places the measurement location close to the pressure antinode for the longitudinal component of this mode.

Conclusions

Jets emanating from mono-injector nozzles were seen to breakup under both tangential mode and mixed longitudinal/tangential acoustic modes into both spray-like and fan-like structures for physical conditions that place the jets in the Rayleigh breakup region. No effect of these modes on the gross atomization characteristics of co-axial nozzles were observed. However, calculations indicate that trajectories of droplets are affected by acoustic oscillations and it seems that measurements of droplet size and velocity in lieu with the phase angle of acoustic oscillations will unearth additional information.

B. Diagnostic Development

As a complementary effort to the atomization and spray studies described previously, a diagnostic development program has been initiated to support the measurement needs of this project.

Spray and droplet measurements in liquid rocket engines require non-intrusive

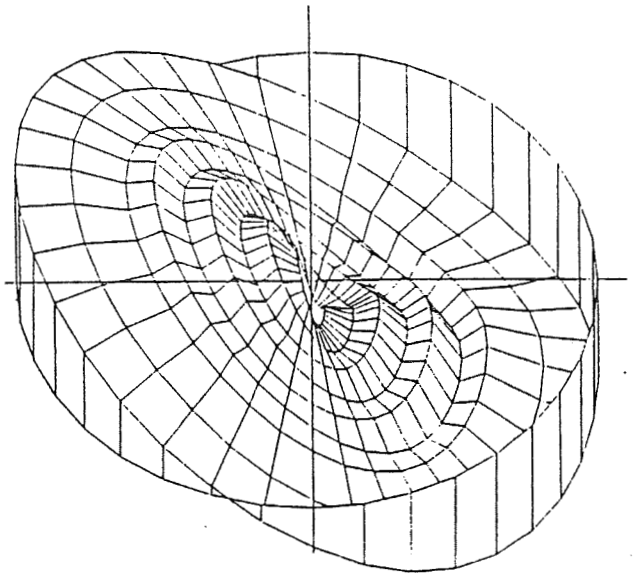


Fig. 2. Spray type breakup observed at a frequency of 1140 Hz. The inset graph shows the pressure amplitude at a horizontal plane (within the chamber) 10 inches from the top of the chamber at zero phase angle. This mode is identified to be the first tangential mode.

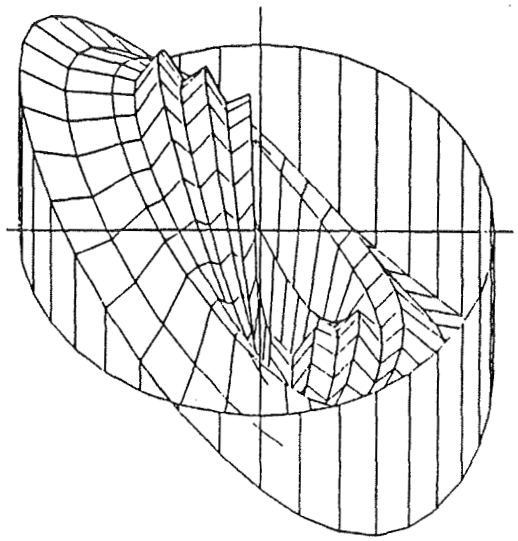


Fig 3. Fan type breakup observed at a frequency of 1560 Hz. The inset graph shows the pressure amplitude at a horizontal plane (within the chamber) 10 inches from the top of the chamber at zero phase angle. Note the high pressure gradient at the center vertical plane. This mode could be the third longitudinal/first tangential mode.

techniques which possess both high spatial and temporal resolution capabilities. In addition, due to the short duration and highly transient nature of liquid rocket combustion processes, techniques which provide extensive spatial measurement capabilities are highly desirable. Recent progress in the development of planar laser imaging techniques have demonstrated the potential for achieving such measurements. In the present work, a planar laser imaging approach for droplet and particle sizing which incorporates an optical polarization ratio technique has been investigated. A series of pressure atomized sprays have been imaged to obtain droplet size measurements based on the ratio of horizontally to vertically polarized scattered light. This technique, which provides an ensemble averaged droplet diameter, provides spatially and temporally resolved droplet size measurements over an extensive spatial region.

Experimental Setup

The polarization ratio approach utilized in these studies involves a pulsed Nd-YAG laser operating at the 532 nm laser line. This laser beam is formed into a sheet of light whose polarization orientation is adjusted using a half-wave plate to provide both vertically and horizontally polarized light components. For the water sprays studied, the polarization vector was oriented at a 45° angle with respect to the horizontal plane of polarization. The images of scattered light intensity at a 90° scattering angle were simultaneously obtained for each polarization orientation using a pair of 35 mm cameras with the aid of an appropriate beam splitter and polarization filter. The images were recorded on black and white film and were subsequently digitized using a CID solid state camera. These digitized images were stored on a personal computer and two-dimensional polarization ratio fields were constructed from the ratio of a pair of vertically and horizontally polarized scattered light images. The ratio of horizontal and vertical polarization intensity can be directly related to appropriate optical scattering cross sections (C_{HH}/C_{VV}). Droplet size information can be directly calculated from MIE light scattering theory. Calculations of these polarization ratio values were utilized to provide droplet size information throughout the spray.

Results

In the present studies, a spray region, 5 cm x 5 cm, was imaged and analyzed. Droplet sizes based on D_{32}, in the water spray were observed to vary between 10 μm in the central region of the spray to 40 μm near the edges of the spray. Comparisons with preliminary measurements utilizing a phase doppler particle sizing approach consistently show a systematic difference in size. The polarization ratio approach typically results in droplet sizes a factor of two to four smaller than the phase doppler results. In addition to providing a sizing capability, the present work demonstrates that the polarization ratio approach can be used to provide a means to discriminate regions containing soot particles from droplet regions. Such a capability will be useful in studies in liquid hydrocarbon rocket combustors where both soot particles and droplets are likely to be present.

III. Proposed Work for Coming Year

Work during the next year will concentrate on performing similar acoustic experiments in an environment that more closely matches actual rocket motors. These experiments will be carried out in a pressurized environment using the current acoustic chamber which can be operated at pressures up to 300 psi. Three types of nozzles will be used: a series of mono-injector nozzles ranging in diameter from 0.0625 inches to 0.1 inches with an aspect ratio of at least 10, a co-axial injector having an inner diameter of 0.1 inches and an outer diameter of 0.2 inches with an aspect ratio of 35, and an impinging nozzle yet to be desiged. Based on past experience, the acoustic modes of interest with the current chamber will be the first tangential mode (1-T mode) at 1140 Hz and the previously discussed first tangential, third longitudinal mode (1-T/3-L mode) at 1560 Hz.

The acoustic drivers will be upgraded to provide a greater pressure field intensity in order to allow other standing wave modes to emerge, such as radial modes. Present observations have shown that as the jet velocity increases, the effect of the applied acoustic field on the spray diminishes. By using more robust acoustic drivers, the mean jet velocity can be increased with the instability effect remaining. In addition, fluids such as Freon 112 and Freon 113 will be used to simulate actual rocket motor fuels and oxidizers.

In terms of measurement techniques, a variety of approaches will be employed including microphone probes for characterizing the induced pressure field, while flash photography, high-speed cinematography, a two-dimensional laser polarization ratio technique, and a laser-based phase doppler particle analyzer will be used to characterize any observed spray instabilitiy phenomena. The photographic techniques yield a global view of the interaction between the applied acoustic field and the injected fluid, whereas the laser techniques allow for quantitative measurements of the observed spray phenomena.

A two-dimensional polarization ratio technique is currently being developed to acquire droplet size data under transient combustion conditions. This technique is based on a thin sheet of laser light emanating from a frequency doubled Nd:YAG pulse laser illuminating a spray. The two-dimensional polarization ratio technique determines the droplet diameter in the plane of the laser sheet using the measured polarization ratio of the scattered laser light from spherical liquid droplets in the spray. In essence, the polarization ratio technique gives spatially extensive droplet size data with high temporal resolution.

The phase doppler particle analyzer uses an Ar-ion laser to maintain a non-intrusive probe volume in a given environment and provides a point measurement of the droplet size in the spray. The phase doppler particle analyzer will continuously record droplet size and velocity distributions inside the probe volume with respect to the phase of the acoustic oscillation.

Thus by using the above two laser diagnostic techniques, a quantitative determination of the effect of the induced pressure field on the spray can be obtained. The techniques provide complementary information and measurement verification concerning the evolution of the spray in the oscillating pressure field.

Concurrent with the continued acoustic instability work in the current chamber, a new transparent rectangular chamber will be constructed. One of the difficulties with the current cylindrical chamber is its 'rich' frequency characteristics. By fabricating a chamber in which the consecutive standing wave frequencies are equally spaced, one can easily locate acoustic modes of interest with a standard frequency generator. The transparent nature of the new cell will allow more optical and visual access to utilize the proposed measurement techniques. In addition, the new cell will allow easier access for microphone probes.

The objective of these experiments will be to better understand the interaction between an oscillating pressure field and the atomization and spray phenomena in rocket combustors. Continued attention will be given to the study of the observed coupling between the pressure field and the breakup of liquid jets as described previously in the results section. However, additional attention will be given to the study of droplet-droplet interactions such as coagulation and secondary breakup which may follow the atomization process. These measurements will be conducted under conditions similar to the liquid break-up studies and will concentrate on quantitative droplet size measurements using the described laser-based techniques.

PRESENTATION 2.2.6

LIQUID JET BREAKUP

AND

ATOMIZATION IN ROCKET CHAMBERS

UNDER DENSE SPRAY CONDITIONS

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LIQUID JET BREAKUP AND ATOMIZATION IN ROCKET CHAMBERS UNDER DENSE SPRAY CONDITIONS

Kenneth K. Kuo, Fan-Bill Cheung,
Roger D. Woodward, and Kenneth N. Garner
NASA Center for Space Propulsion Engineering

I. Research Objectives and Potential Impact on Propulsion

One of the most important issues in liquid rocket propulsion is how to promote the mixing of liquid propellants and eliminate the problem of combustion instability due to inadequate spays in thrust chambers. To address this issue, it is necessary to seek a good understanding of important mechanisms controlling the liquid jet breakup and atomization processes in the dense spray region. Owing to the complexity of the phenomena and the difficulty in obtaining experimental measurements, previous studies of spray combustion have focused largely on the dilute region, as identified in the 1988 Workshop on "Mixing and Demixing Processes in Multiphase Flows with Applications to Propulsion Systems," sponsored by the University Space Research Association of NASA Marshall Space Flight Center and the 1989 JANNAF Workshop coordinated by Chiu and Gross. Very little work has been conducted to explore the characteristics of dense spray, including the behavior of the jet boundary surface, core breakup length, liquid jet ligament size, local void fraction, and velocity distribution. In the absence of these data, it is impossible to realistically predict the rates of mixing of liquid propellants and thus engine performance.

To fill the above technological gaps and to extend the state-of-the-art in spray combustion, the present research project was initiated in September 1988 to study the phenomena of dense spray. Two advanced diagnostic techniques have been established and employed in the project. The first technique involves the use of a real-time X-ray radiography system along with a high-speed CCD Xybion camera and an advanced digital image processor to investigate the breakup processes of the liquid core. The focus of this part of the project is to determine the inner structure of the liquid jet and to correlate the core breakup length and local void fraction to various controlling parameters such as the characteristic Reynolds and Weber numbers. The second technique involves the use of a high-power copper-vapor laser to illuminate the liquid jet via thin sheets of laser light, with the scattered light being photographed by a Xybion electronic camera synchronized to the laser pulse. This technique, which is capable of recording the breakup event occurring within 25 nano-seconds, enables us to freeze the motions of the jet and liquid droplets. The focus of this part of the project is to determine the outer structure of the liquid jet and to discover the configuration of the surface waves, the spray pattern, and the droplet size distribution in the non-dilute region. Results obtained by these two advanced diagnostic techniques will provide the much needed database for model development and accurate prediction of engine performance. The present work also represents a breakthrough in the area of advanced diagnostics of dense sprays.

II. Current Status and Results

To overcome the difficulty in experimental measurements in the dense spray region, the aforementioned advanced diagnostic techniques (i.e., the real-time X-ray radiography and the laser-illuminated flash-photography techniques) have been established and employed in the project. A liquid injection system was designed and fabricated for this purpose. The injector consists of a single coaxial element of inner jet diameter and annular area similar to that of a single SSME injector element. The injector used in this project is of modular design so that the inner jet diameter and annular gap dimensions can be easily varied by exchanging appropriate components. The fluid supply and recovery system was assembled in conjunction with the X-ray radiography system, and a number of real-time radiography tests was performed to calibrate the test procedure and working fluid.

Several flash-photographic tests of coaxial jets at various conditions were conducted using light-pulse illumination with a pulse duration of ~10 μs. Unfortunately, this pulse duration was found to be too long to completely freeze the liquid droplet/ligament motions for all but the lowest injected gas-to-liquid velocity ratios. Nevertheless, the series of photographs did provide useful qualitative information on the physical mechanisms governing the liquid jet breakup due to the coaxial gas flow. To upgrade the flash-photography technique, a copper-vapor laser and a CCD Xybion electronic camera were obtained recently. The laser, with a pulse duration of ~30 ns, and the camera, with a gated exposure time ranging from 25 ns to 50 ms, can be synchronized to produce a video tape of the jet breakup event consisting of a series of completely-frozen-motion pictures. The laser beam delivery system and the optical setup for laser sheet generation are currently under construction.

Several real-time X-ray radiography tests of coaxial jets have been successfully conducted using nitrogen gas and an X-ray absorbing aqueous solution of potassium iodide as the working fluids. The mass flow rates of liquid and gas have been calibrated, and were recorded for each test to determine the relative velocity used to form the characteristic Reynolds and Weber numbers for qualitative analysis of data. A typical run involves passing a continuous stream of X-rays through a section of the near-injector region of the coaxial jet. Where the liquid fraction is highest, the greatest amount of X-ray attentuation occurs. The X-rays that reach the screen of the image intensifier are converted to light photons, and an optical signal is received by the Xybion electronic camera. The output from the camera is in RS-170 video format consisting of 30 controlled-exposure pictures per second, each picture having an exposure time as short as 25 ns. The video signal is then recorded on tape and analyzed using the Quantex 9210 digital image processor to determine the liquid jet inner structure, core breakup length, and local void fraction for each frame of the video sequence.

A typical set of results for coaxial jets obtained from the image analysis of the real-time X-ray radiography tests is depicted in Figures 1 and 2. Figure 1a shows a single frame from the video sequence of a coaxial jet after an analysis procedure that equalizes the X-ray image in the selected area. In this procedure, the regions of highest liquid fraction in the area are assigned a radiance level of zero, and the

highest void regions assigned a radiance level of 255. This spreads out the gray scale across the area of interest for more pronounced distinction of the image, thus greatly enhancing direct visualization of the jet. Figures 1b to 1d show the radiance level distributions across the jet in both vertical (i.e., transverse) and horizontal (i.e., streamwise) directions. Figure 1b is a vertical profile across the jet taken at an upstream location where there is still a well-defined liquid core. On the other hand, Fig. 1c is a vertical profile taken at a location downstream of the core breakup length. At this location, the liquid core is no longer intact as voids are evident within the core region. Figure 1d is a horizontal profile along the jet centerline. The large void in the core depicted by the spike in the radiance profile clearly indicates that the core is no longer intact. Figure 1e is an example of the isophote analysis in which regions falling into a selected range of radiance levels are shaded. In this figure, the darkest region includes the liquid ligaments and large droplets, whereas the shaded region represents the intact liquid core. Based on this isophote result, the core breakup length for this low-pressure test (@ 1 atm) was determined to be about 5-3/4 inches from the injector. A similar isophote picture for a coaxial jet at higher gas-to-liquid relative velocity is shown in Fig. 1f, where the core breakup length was found to be about 3-1/4 inches. The decrease in the breakup length is evidently due to an increase in the characteristic Reynolds number. Figures 2a and 2b demonstrate the capability of the image processor to assign color values to the different shades of gray, allowing human eyes to better distinguish the existence of various regions. Figure 2a is from the test case of that depicted in Fig. 1e whereas Fig. 2b corresponds to Fig. 1f. Note the difference in core breakup length for the two cases, where the continuous blue color region represents the intact liquid core. Finally, Figs. 2c and 2d show another image analysis feature (i.e., the "zoom") that magnifies the X-ray image, allowing a close-up observation of the details of the liquid core and ligament formation region. Quantitative analysis of the data is now underway.

The above X-ray results represent a set of benchmark data for liquid jet breakup in the dense spray region that has not been observed heretofore. These data, together with those to be obtained in the coming year, will provide a much needed database for the development and validation of liquid-rocket-engine performance models.

III. Proposed Work for Coming Year

The X-ray radiography study of coaxial jet breakup in the non-dilute region under open-atmosphere conditions will be completed in the very near future. More results similar to those shown in Section II will be obtained so that the liquid core breakup length can be correlated to the characteristic Reynolds and Weber numbers as well as other pertinent parameters such as the injector flow area ratio (i.e., ratio of the annular gas flow area to liquid flow area). With the modular design of the injector, this area ratio can be easily varied. To complete the liquid jet breakup study for the case of injection into atmospheric pressure, the laser-assisted flash-photography technique will also be used to produce stop-action video pictures of the coaxial jets with very short exposure times (~25 ns). A data correlation for the surface breakup phenomena in the non-dilute region will be performed.

In addition to the above work items, much of the effort in the coming year will be devoted to studying the breakup processes of a coaxial flow injected into a high-pressure chamber in order to simulate more closely the liquid rocket engine environment. An existing high-pressure, windowed test chamber will be modified for this purpose. The chamber has a large flow area for the spray to develop with full length windows available for the lower pressure tests (<100 psig). Smaller windows will be fabricated for tests up to about 1,000 psig pressure. The pressure in the chamber, to be monitored with a pressure transducer, will be held constant during the tests by the use of a back-pressure regulator. A bursting diaphragm will be included as a safety precaution in case of overpressurization. Also, a liquid collection reservoir will be added so that the liquid will not fill up the test area. This reservoir may serve as a surge tank as well. The pressurized windowed test chamber will be employed for both real-time X-ray radiography and laser-assisted flash-photography studies. In the former case, the image processing technique will be used whereas in the latter case, a secondary window will be machined at the top of the chamber to direct the laser sheet into the test chamber in order to illuminate the jet. Video data analogous to those obtained in the open atmosphere tests will be acquired for various chamber pressures. The correlations deduced from the one atmosphere data will be modified and extended to include the effect of variations in chamber pressure. Additional open-atmosphere and high-pressure tests will be performed to further confirm the validity of the correlations. Results including the effect of elevated chamber pressures will be obtained, which can be directly applicable to liquid rocket engines. They will provide a useful guideline for modeling the jet breakup processes as well as serve as an empirical input into engine performance models. A detailed itemized work statement for the coming year is given below.

1. Complete the X-ray radiography study of coaxial jet breakup under open-atmosphere conditions.

2. Correlate liquid core breakup length based on the open-atmosphere data.

3. Conduct laser-assisted flash-photography tests under open-atmosphere conditions.

4. Modify the existing windowed test chamber for high-pressure studies.

5. Calibrate flow conditions inside the pressurized chamber at different operating conditions.

6. Conduct X-ray radiography jet-breakup tests in the pressurized chamber.

7. Conduct laser-assisted flash photography tests in the pressurized chamber.

8. Extend the open-atmosphere correlations to include elevated pressure effect.

9. Perform additional open-atmosphere and high-pressure tests to further confirm the validity of the correlations.

10. Initiate model formulation of the two-phase coaxial jet breakup processes.

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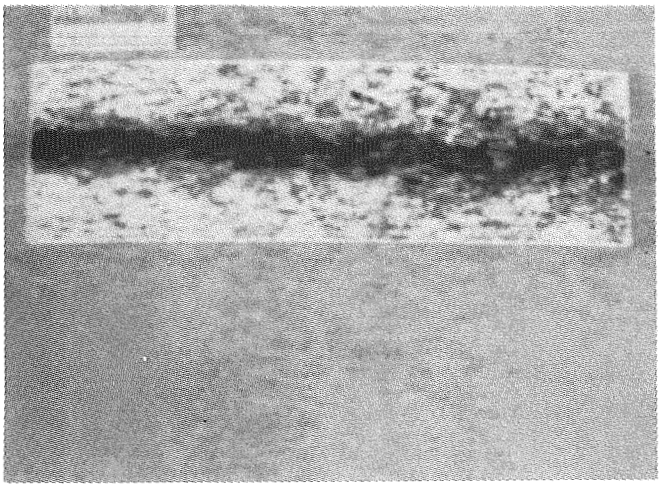


Figure 1a. X-ray image of coaxial jet after equalization (contrast stretching). Figures 1b-1e are derived from this image.

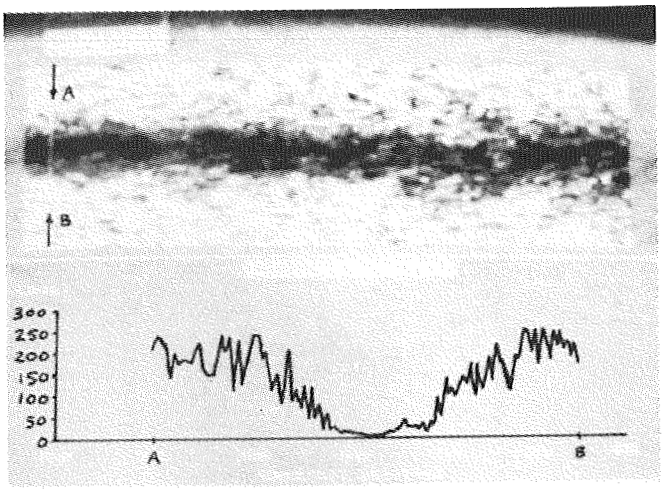


Figure 1b. Profile of radiance level versus vertical position across the jet at a location approximately 2½ inches downstream of the injector. Arrows indicate location where profile was taken. Profile indicates well-defined liquid core.

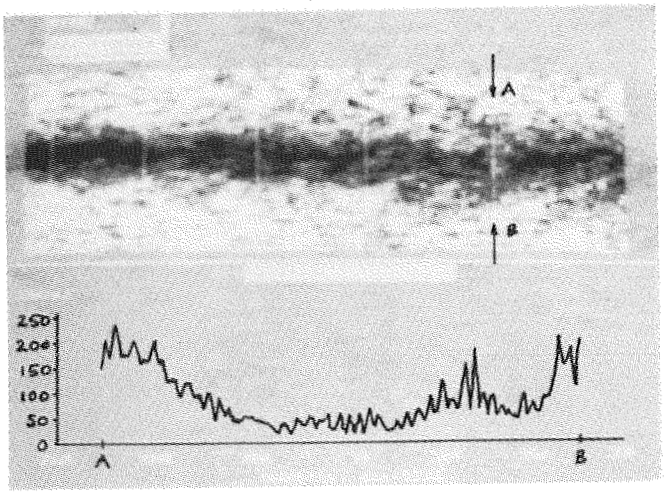


Figure 1c. Profile of radiance level versus vertical position across the jet at a location downstream of the liquid core breakup length. Arrows indicate location where profile was taken.

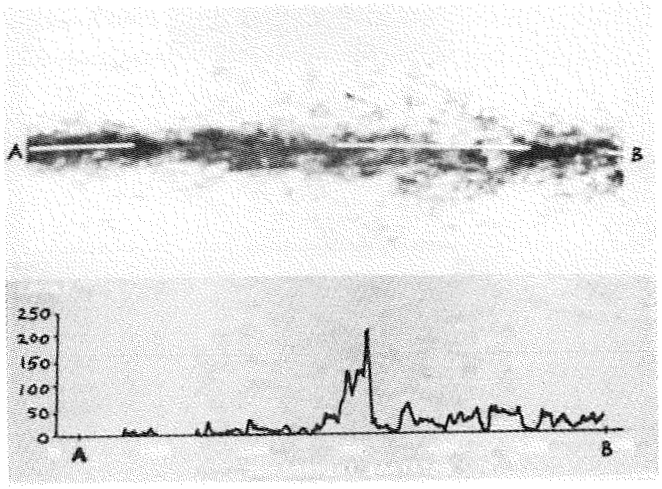


Figure 1d. Profile of radiance level versus horizontal position along the jet centerline. The spike in the radiance profile indicates that the liquid core is no longer intact at that location.

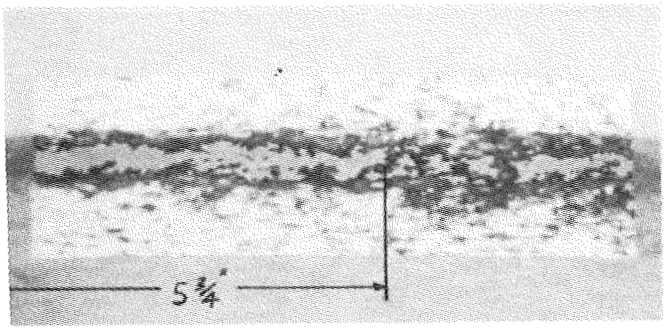


Figure 1e. Isophote analysis for coaxial jet. Same jet as depicted in Figures 1a-1d. Core breakup length is indicated at about 5 3/4 inches from the injector (1 atm test).

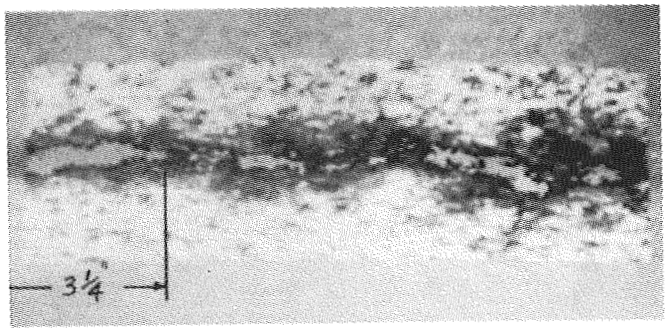


Figure 1f. Isophote analysis for a coaxial jet of higher injected gas-to-liquid relative velocity (1 atm test). Note the decrease in indicated core breakup length to 3¼ inches from the injector.

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PRESENTATION 2.2.7

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DROPLET-TURBULENCE INTERACTIONS IN SUBCRITICAL AND SUPERCRITICAL EVAPORATING SPRAYS

DROPLET-TURBULENCE INTERACTIONS IN SUBCRITICAL AND SUPERCRITICAL EVAPORATING SPRAYS

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Edward Coy
Stuart Greenfield
Young-Hoon Song
Mechanical Engineering Department

I. Research Objectives and Potential Impact on Propulsion

The objective of this research is to obtain an improved understanding of droplet-turbulence interactions in vaporizing liquid sprays under conditions typical of those encountered in liquid fueled rocket engines. The interaction between liquid droplets and the surrounding turbulent gas flow affects droplet dispersion, droplet collisions, droplet vaporization and gas-phase, fuel-oxidant mixing, and therefore has a significant effect on the engine's combustion characteristics. An example of this is the role which droplet-turbulence interactions are believed to play in combustion instabilities. Despite their importance, droplet-turbulence interactions and their effect on liquid fueled rocket engine performance are not well understood. This is particularly true under supercritical conditions, where many conventional concepts, such as surface tension, no longer apply. Our limited understanding of droplet-turbulence interactions, under both sub-critical and supercritical conditions, represents a major limitation in our ability to design improved liquid fueled rocket engines. It is expected that the results of this research will provide previously unavailable information and valuable new insights which will directly impact the design of future liquid fueled rocket engines, as well as, allow for the development of significantly improved spray combustion models, making such models useful design tools.

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II. Current Status and Results

The primary efforts to date have been devoted to the development of the experimental apparatus and diagnostic techniques required for this study. This includes the development of a flow system which is capable of simulating the broad range of turbulent flow conditions encountered in the peripheral regions of coaxial and impinging type rocket sprays. It is in this region (see Figure 1) where droplet-turbulence interactions are most important and have significant effects on droplet vaporization, droplet dispersion and droplet collisions, as well as, on gas-phase, fuel-oxidant mixing. The basic concept of this flow system is illustrated in Figure 2. A novel turbulence generator [1] is used to produce turbulent flow conditions which are uniform over the cross-section of the test section to within ±10%, with relative turbulence intensities of up to 70% and mean velocities of up to 50 m/sec. A uniform, 10mm diameter, polydispersed spray of variable droplet density and size distribution is produced using a low pressure spray nozzle and skimmer combination, as illustrated in Figure 2. Droplet size distributions between 10 and 200 microns, droplet number densities between 10^3/cc and 10^4/cc, and mean drop velocities of up to 10m/sec. can be achieved with this device. This spray is transversely injected into the one-dimensional turbulent flow, thereby providing a well defined region of droplet turbulence interactions. A single droplet generator is also being developed in order to study the interaction between individual droplets and turbulence.

An atmospheric pressure, room temperature version of this system is currently in operation and has been used extensively for diagnostic development. A high pressure (70 atm) elevated temperature (300°C) turbulent flow system has also been designed and is currently being fabricated. This system will be capable of achieving supercritical conditions for a number of liquids including liquid oxygen, liquid nitrogen, as well as most liquid hydrocarbons. A schematic drawing of this system is shown in Figure 3, along with the single droplet generator which is being fabricated for use in this study.

Droplet-turbulence interactions basically refer to the mass, momentum and energy transport processes which occur between individual droplets and the surrounding turbulent gas. This interaction is important both under conditions of low droplet number density, where the droplets can be treated as individual, isolated droplets, as well as under conditions of high droplet number density, where droplet-droplet interactions become important. In order to characterize the exchange of mass, momentum and energy between the droplets and the turbulent gas flow it is necessary to measure the properties of individual droplets (i.e. size, velocity, temperature) and of the gas (i.e. mean velocity, turbulence intensity, integral scales, energy spectrum, and fuel/oxidant). In the case of the droplets, since the transport rates depend on the droplet size, it is necessary to simultaneously measure the size and velocity, for example, of individual droplets in order to characterize droplet drag. The desired size-velocity and size-temperature correlations can be obtained with point measurement techniques, however, it was decided to use two-dimensional, laser sheet imaging which also provides information on the spray structure. The technique which has been developed for these measurements is referred to as double-pulsed, laser sheet, fluorescence imaging [2]. This technique is

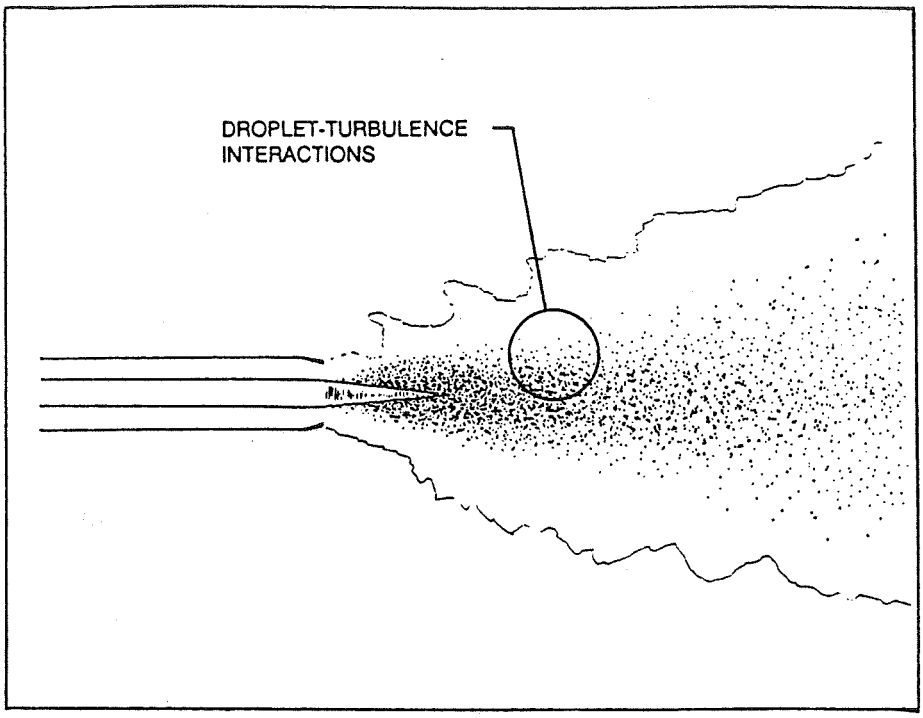


Figure 1. Region of droplet-turbulence interactions in co-axial liquid rocket spray.

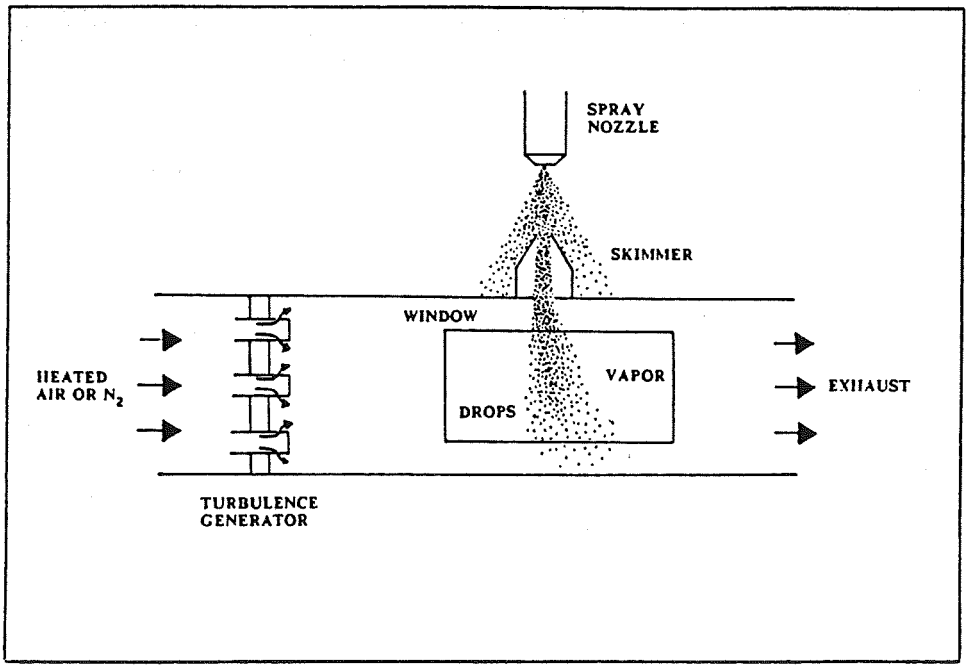


Figure 2. Turbulent flow system for study of droplet-turbulence interactions.

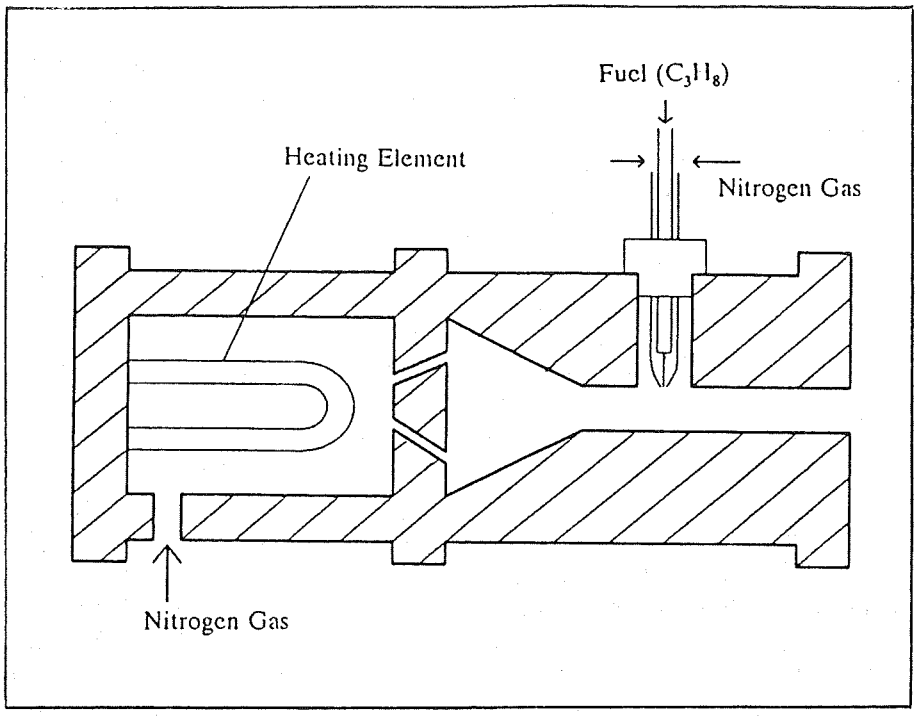


Figure 3. High pressure, high temperature turbulent flow system for study of droplet-turbulence interactions under supercritical conditions.

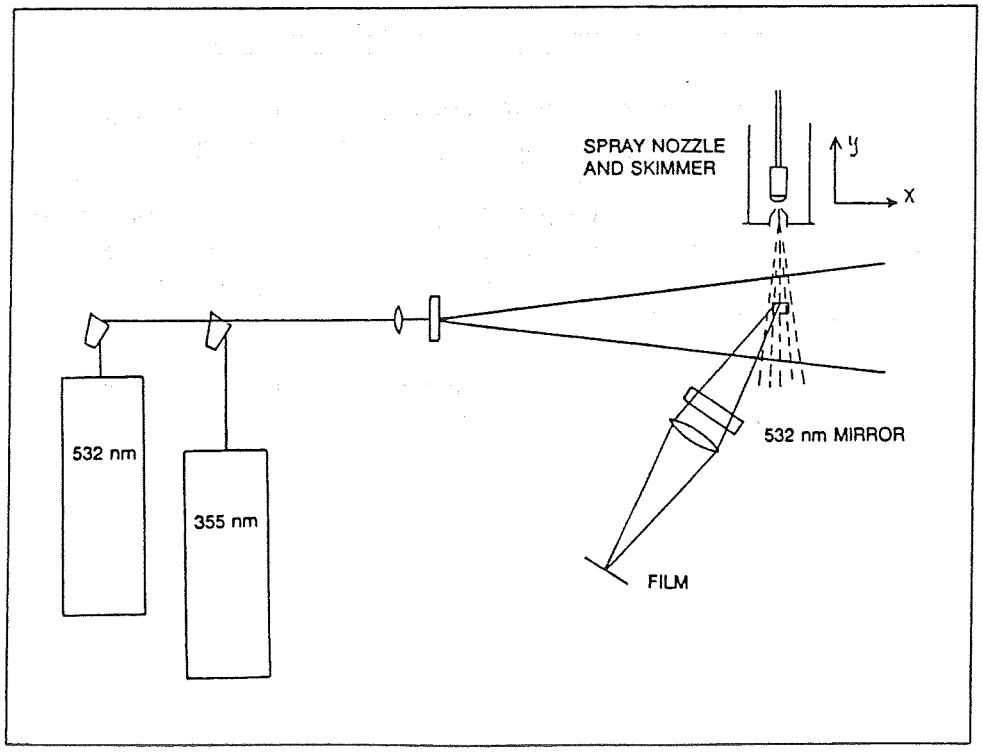


Figure 4. Double pulse, laser sheet, fluorescence imaging technique.

illustrated in Figure 4. It involves the use of two pulsed Nd:YAG lasers which are focused to form two coincident laser sheets of approximately 500 micron thickness which pass through the spray. The two lasers are synchronized, but with a variable delay (e.g. 100 microseconds) between the two pulses. The first laser operates at 532 nm and the second at 355 nm. Two fluorescent dyes are added to the spray liquid, one which is excited by the 532 nm laser sheet and produces red fluorescence and the second which is excited by the 355 nm laser sheet and produces blue fluorescence. The droplet images are then recorded on color slide film, through a 532 nm mirror which eliminates the Mie scattering, and appear as red and blue droplet pairs. The spacing between the droplets defines the x and y components of the droplet velocity and the size of the droplets are determined directly from the image. In addition, one of the dyes, i.e. the one excited at 355 nm, is what is referred to as an exciplex [3]. Exciplex fluorescence has a number of unique characteristics depending on the dye concentration. One of these is that the ratio of the fluorescence intensity at two different wavelengths, e.g. 400 nm and 500 nm, is linearly proportional to the droplet temperature. Combining these techniques, provides simultaneous measurements of the size, velocity and temperature of individual droplets. A PC-controlled stepping motor and image analysis system is then used for automated signal processing of such images. The data can then be examined in a number of different formats, e.g. droplet velocity distribution versus droplet size, from which quantitative information on droplet drag and vaporization rate can be determined. It is also important to characterize the effect of the droplets on the turbulence properties of the surrounding gas. In order to do so, laser Doppler velocimetry is used to obtain measurements of the mean velocity, turbulence intensity, integral scales and energy spectrum.

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SECTION 2.3

LIQUID PROPULSION TECHNOLOGIES

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PRESENTATION 2.3.1

N91-28229

COMPOSITE MATERIAL SYSTEMS FOR HYDROGEN MANAGEMENT

COMPOSITE MATERIAL SYSTEMS FOR HYDROGEN MANAGEMENT

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Dept. of Engineering Science and Mechanics

I. Research Objectives and Impact on Propulsion

The task of managing hydrogen entry into elevated temperature structural materials employed in turbomachinery is a critical engineering area for propulsion systems employing hydrogen or decomposable hydrocarbons as fuel. Extant structural materials, such as the Inconel series, are embrittled by the ingress of hydrogen in service, leading to a loss of endurance and general deterioration of load-bearing dependability. Although the development of hydrogen-insensitive material systems is an obvious engineering option, to date insensitive systems cannot meet the time-temperature-loading service extremes encountered. A short-term approach that is both feasible and technologically sound is the development and employment of hydrogen barrier coatings.

The present project is concerned with developing, analyzing, and physically testing laminate composite hydrogen barrier systems, employing Inconel 718 as the structural material to be protected. Barrier systems will include all metallic, metallic-to-ceramic, and, eventually, metallic/ceramic composites as the lamellae. Since space propulsion implies repetitive engine firings without earth-based inspection and repair, coating durability will be closely examined, and testing regimes will include repetitive thermal cycling to simulate damage accumulation. The target accomplishments include:

a. Generation of actual hydrogen permeation data for metallic, ceramic-metallic, and hybrid metallic/ceramic composition barrier systems, practically none of which is currently extant.

b. Definition of physical damage modes imported to barrier systems due to thermal cycling, both transient temperature profiles and steady-state thermal mismatch stress states being examined as sources of damage.

c. Computional models that incorporate general laminate schemes as described above, including manufacturing realities such as porosity, and whatever defects are introduced through service and characterized during the experimental programs.

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II. Current Status and Results

Efforts to date have been directed towards the following three areas:

A. Laminate system modelling
Before designing and engaging in relatively complex physical determination of the degree of hydrogen permeation in laminate systems, numerical models were developed to ascertain order-of-magnitude throughput of hydrogen gas. Approximate answers can only be calculated at this time, because permeation data is only available for selected materials, and then often only for bulk samples, not typical of films and coatings. For the case of solid and porous ceramic films, for example, there are orders of magnitude greater ingress (and egress) of hydrogen into Inconel 718, as can be seen in the following figure.

B. Thermodynamics of Stress States and Permeation
Recent theoretical developments in thermodynamics hypothesize that permeation of hydrogen is proportional to the hydrostatic, or mean, component of normal stress. However, experimental characterization of material response parameters in this regard are sparse, and do not pertain to relevant materials where they are extant. At the same time, even uniform temperature distributions will lead to residual stress generation in laminate systems, due to thermal mismatch of layer constituents. Coarse extrapolation of marginally relevant data on stress dependence of permeation, matched to realistic levels of stress generated by typical service temperature ranges, strongly suggests that permeation can be measurably altered in real proposed material systems.

C. Experimental Design
Parts (A) and (B), above, have driven the particulars of a permeation test cell design that is complete and being fabricated at the time of this writing. At the extreme of laminate effectiveness represented by fully dense ceramic barrier coatings (hardly realizable in practice, but approachable to some degree), downstream vacua levels of 10^{-9} Torr were indicated. Accordingly, the permeation cell design was necessarily of the ultrahigh vacuum type (UHV), and a mass spectrometer was deemed necessary to measure the (possibly) slight levels of hydrogen egressing from system samples during the transient phase.

In-situ heating of composite systems was dictated by surface cleanliness considerations whenever temperature effects were to be examined. Temperature was deemed crucial both for residual stress generation as well as fatigue damage generation during service cycling. A quartz lamp heat source and light wire assembly, with suitable UHV transparent porting was designed into the permeation cell system.

At the present time, the cell is being assembled in Room 22 of the Research Building East, the Center headquarters.

III. Proposed Work for Coming Year

A. Experimental facilities will be completed by midsummer, 1990. Completion implies that the UHV hardware will be assembled and tested successfully, including all instrumentation. Two test cells are being manufactured, and their incorporation is expected by early August.

B. Composite material system fabrication will be begun in July as well. This phase of the project will be ongoing for the entire project duration. Various coating technologies will be evaluated to provide metallic, ceramic, and composite coatings.

C. Experimental testing will begin towards the end of August. Several goals will drive the scheduling of the experimental program.

1. Permeation data for many of the candidate coating materials does not exist and will have to be established. In addition, the physical structure of coatings is often vastly different from that associated with monolithic samples of the same chemical composition. Accordingly, permeation data will be obtained for coating microstates that are relevant and related to the particulars of the achievable microstates.

2. Downstream hydrogen concentrations in a given coating will determine availability of the gas to the next constituent in the laminate. That, and the particulars of the laminate interface, will strongly affect multi laminate permeation and this type of system study will be necessary for input into design modelling schemes.

3. Designed residual stress states will be engineered into laminate systems by controlling the choice of thermal response of constituents, by controlling fabrication time/temperature histories, and, finally, by specification of testing (service simulation) temperature.

4. Finally, studies will be undertaken to assess the damage imparted by the thermal cycling that is endemic to space propulsion engine cycling.

It is anticipated that items 1 through 4 will progress throughout the entire year. Although substantial progress towards completion should be accomplished during that time interval, it is anticipated that followup project funding from outside sources will be required to finish even these limited objectives.

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PRESENTATION 2.3.2

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ROBUST AND REAL-TIME CONTROL OF MAGNETIC BEARINGS FOR SPACE ENGINES

ROBUST AND REAL-TIME CONTROL OF MAGNETIC BEARINGS FOR SPACE ENGINES

by
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1. RESEARCH OBJECTIVES AND POTENTIAL IMPACT ON PROPULSION

The rotating machines used in space engines; e.g., orbital transfer vehicles, space shuttle main engines (SSME), operate at much higher speeds compared to those used in ground based engines or aircraft engines. The SSME oxygen turbopumps operate at about 30,000 rpm; and rotational speeds of space engines can be as high as 100,000 rpm. At these high speeds, conventional bearings; e.g., ball bearings, are not reliable. Consequently the main objective of this research program is to develop a highly reliable magnetic bearing system for space engine applications.

In recent years, a number of researchers have developed magnetic bearing systems. However, their main focus has been the applications to relatively low speed engines. Currently, NASA Lewis Research Center is developing magnetic bearings for SSME turbopumps. The control algorithms which have been used are based on either the proportional-intergral-derivative control (PID) approach or the linear quadratic (LQ) state space approach. These approaches lead to an acceptable performance only when the system model is accurately known, which is seldom true in practice. For example, the rotor eccentricity, which is a major source of vibration at high speeds, cannot be predicted accurately. Furthermore, the dynamics of a rotor shaft, which must be treated as a flexible system to model the elastic rotor shaft, is infinite dimensional in theory and the controller can only be developed on the basis of a finite number of modes. Therefore, the development of the control system is further complicated by the possibility of closed loop system instability because of residual or uncontrolled modes, the so called spillover problem. Consequently, novel control algorithms for magnetic bearings are being developed to be robust to inevitable parametric uncertainties, external disturbances, spillover phenomenon and noise. Also, as pointed out earlier, magnetic bearings must exhibit good performance at a speed over 30,000 rpm. This implies that the sampling period available for the design of a digital control system has to be of the order of 0.5 milli-seconds. Therefore, feedback coefficients and other required controller parameters have to be computed off-line so that the on-line computational burden in extremely small.

The development of the robust and real-time control algorithms is based on the sliding mode control theory. In this method, a dynamic system is made to move along a manifold of sliding hyperplanes to the origin of the state space. The number of sliding hyperplanes equals that of actuators. The sliding mode controller has two

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parts: linear state feedback and nonlinear terms. The nonlinear terms guarantee that the system would reach the intersection of all sliding hyperplanes and remain on it when bounds on the errors in the system parameters and external disturbances are known. The linear part of the control drives the system to the origin of state space. Another important feature is that the controller parameters can be computed off-line. Consequently, on-line computational burden is small.

2. CURRENT STATUS AND RESULTS

2.1 CONTROL ALGORITHM FOR RIGID ROTOR

First, the flexibility of the rotor shaft has been ignored and the basic understanding of the bearing control system has been obtained. The equations of motion for the generic model of a rotor shaft (Figure 1) have been obtained using Newton and Euler laws. For small displacements, the linearized equations have the following form:

$$\ddot{X} + P\dot{X} = BU + VS_e + D \tag{1}$$

where $X \in R^4$ and represents the radial displacements at locations of magnetic bearings. Elements of 4 x 1 control input vector V correspond to directions of the elements of C. The matrix P represents the gyroscopic effects. The 2 x 1 vector S_e is simply $[\sin \Omega t \quad \cos \Omega t]^T$ where Ω is the angular speed of the shaft. The vector D contains transient disturbance, and it is assumed that upper bound on each element, d_{max}, is known. The uncertainty in the model due to the lack of precise knowledge of the rotor eccentricity is contained in the 4 x 2 matrix V. It is assumed that

$$\left| \mho_{ij} - \hat{\mho}_{ij} \right| \le F_{ij} \tag{2}$$

where $\hat{\mho}_{ij}$ is the estimate of $\mho_{ij}$ and F_{ij} is the maximum error, which is considered to be known.

The sliding mode control law is as follows:

$$(BU)_i = -\lambda_i \dot{x}_i - \hat{\mho}_{i1} \sin \Omega t - \hat{\mho}_{i2} \cos \Omega t - \overline{K}_i \, \mathrm{sat}\left(\frac{S_i}{\phi_i}\right) \tag{3}$$

where

$$S_i = \dot{X}_i + \lambda_i X_i$$
$$\overline{K}_i = F_{i1} |\sin \Omega t| + F_{i2} |\cos \Omega t| + d_{max} + \eta_i$$
$$\phi_i = \overline{K}_i / \lambda_i$$
$$\lambda_i > 0$$

Here, η_i is a small positive number. Note that S_i=0 defines the sliding line (Figure 2) and ϕ_i is the time varying thickness of the boundary layer, which has been introduced around the sliding line to eliminate the chattering behavior.

A computer program has been developed to simulate the digital implementation of the control law, eq. (3). A representative steady state response of the controlled rotor is shown in Figure 3. The effects of λ_i and sampling period T_s on the performance of the control system are being examined.

A novel algorithm has been developed to determine the current in the magnetic bearing so that the control force required by equation (3) will be exactly achieved in spite of the nonlinear relationship among the magnetic force, coil current and the air gap. Using this algorithm, the level of maximum current will be estimated for the parameters of space engines.

2.2. FLEXIBLE ROTOR DYNAMIC MODEL

A mathematical model of the flexible rotor system has been formulated. The rotor system is modeled as a fixed-free axisymmetric shaft with an unbalanced disk inertia, and supported by two electro-magnetic bearings, or four independent actuators (Figure 1). The equations of motion and boundary conditions shown below are derived by applying the Hamilton's principle.

The equations of motion are:

$$mx_{,tt}+EIx_{,zzzz}+(m_d x_{,tt}-I_p\Omega y_{,zzt}-I_t x_{,zztt})\delta(z-d) = F_{13}\delta(z-a)+F_{57}\delta(z-b)+\rho e\Omega^2\cos\Omega t\,\delta(z-d) \quad (4)$$

$$my_{,tt}+EIy_{,zzzz}+(m_d y_{,tt}+I_p\Omega x_{,zzt}-I_t y_{,zztt})\delta(z-d) = F_{24}\delta(z-a)+F_{68}\delta(z-b)+\rho e\Omega^2\sin\Omega t\,\delta(z-d) \quad (5)$$

The boundary conditions are:

$$x(0,t) = x_{,z}(0,t) = y(0,t) = y_{,z}(0,t) = 0 \quad (6)$$

$$x_{,zz}(L,t) = x_{,zzz}(L,t) = y_{,zz}(L,t) = y_{,zzz}(L,t) = 0 \quad (7)$$

Here,

Ω = shaft rotational speed; E = elastic modules; m_d = disk mass
I = shaft second moment of area; m = shaft mass per unit length; L = shaft length
ρ = disk unbalanced mass; e = disk eccentricity; δ = Delta function
I_p = disk polar moment of inertia; I_t = disk transverse moment of inertia
F13 = bearing force F1-F3; F24 = bearing force F2-F4
F57 = bearing force F5-F7; F68 = bearing force F6-F8
(),t = partial derivative with respect to time
(),z = partial derivative with respect to space coordinate

Space engine rotor models developed by other researchers will be examined. The important results from these previous analyses will be used to adjust and tune the above model to reflect the space engine rotor dynamic characteristics. The final model will be used as the basis for the dynamic analysis and control algorithm synthesis of flexible rotors.

2.3 EXPERIMENTAL DESIGN

In parallel to the analytical efforts, an experiment has been planned and designed to validate the rotor model, implement the control algorithm, and verify the theoretical predictions. A rotor fixture from Bently Nevada Company (Figure 4) has been specified and purchased. The fixture consists of a shaft with lumped disk inertia. The unbalanced force of the rotor can be adjusted through insertion of weights into the disks. The shaft speed is controlled by an AC motor. Measurements of the transverse vibration of the rotor are performed by using non-contact displacement probes. Two magnetic bearings are being designed and fabricated by Magnetic Bearing Corporation. The rotor fixture is presently being modified to incorporate the magnetic bearings. A PC-based micro-processor control system has been specified and is being set up as the main controller for the control algorithm implementation.

3. PROPOSED WORK FOR COMING YEAR

3.1 CONTROL ALGORITHM FOR FLEXIBLE ROTOR

The number of vibratory modes of a flexible rotor is infinite in theory and extremely large in practice. Since it is not practical to control vibration in all the modes, the controller will be designed on the basis of a finite number of modes, which is termed as controlled modes. The remaining higher frequency modes are called 'residual modes'. Since magnetic bearings will provide control forces in four independent directions, the effective numbers of actuators will be less than the number of controlled modes, N. In this case, sliding hyperplanes will have to be taken as $S = EX$ where S and X are respectively 4 x 1 and 2N x 1 vectors, and E is a full matrix unlike the situation for the rigid rotor. Furthermore, all the elements of the state vector X cannot be directly measured. Consequently, these states have to be obtained using an observer. The controller design will essentially involve an appropriate choice of the matrix E and the observer gain matrix. Using the singular value robustness tests, these matrices will be chosen such that the closed loop system is asymptotically stable in the presence of residual modes also. Steady state analyses will be performed to determine the influence of matrix E on the magnitude of vibration level and the control force. The objective of this analysis will be to determine an optimal E for given characteristics of magnetic bearings.

The applications of model reference, sliding mode adaptive control technique will also be investigated. In this approach, the sensor output can be made to behave like a response of a reference model having desired damping ratio and natural frequency. The rotor response at those locations where sensors are not mounted will be investigated.

Rotor parameters of space engines will be used in designing these control laws. Various implementation requirements such as power, current etc. will be estimated.

3.2 EXPERIMENTAL VALIDATION OF THE CONTROL SYSTEMS

Two phases of hardware development will be carried out as follows:

Phase I - Test Stand Set Up and System Characterization

The rotor fixture, the magnetic bearings, the sensors and the controller will be integrated and a shakedown test will be performed. A series of tests will first be carried out to determine the magnetic bearings' dynamic characteristics. The rotor will be excited at various positions with impulsive, step, and periodic inputs, in order to characterize the structural dynamics. The results of these efforts will be used to validate the analytical model and modify the control law.

Phase II - Control Implementation and Validation

In the second phase of the experimental study, sensors and actuators for control purposes will be applied to the test stand according to the recommendations from the analytical work. The rigid rotor control algorithm will first be tested. System parameters, such as sensor locations, shaft length, shaft speeds, and rotor unbalanced force, will be varied systematically to examine the performance and robustness of the controller. The results from this phase will provide a quantitative measure of the efficacy of the proposed control strategy.

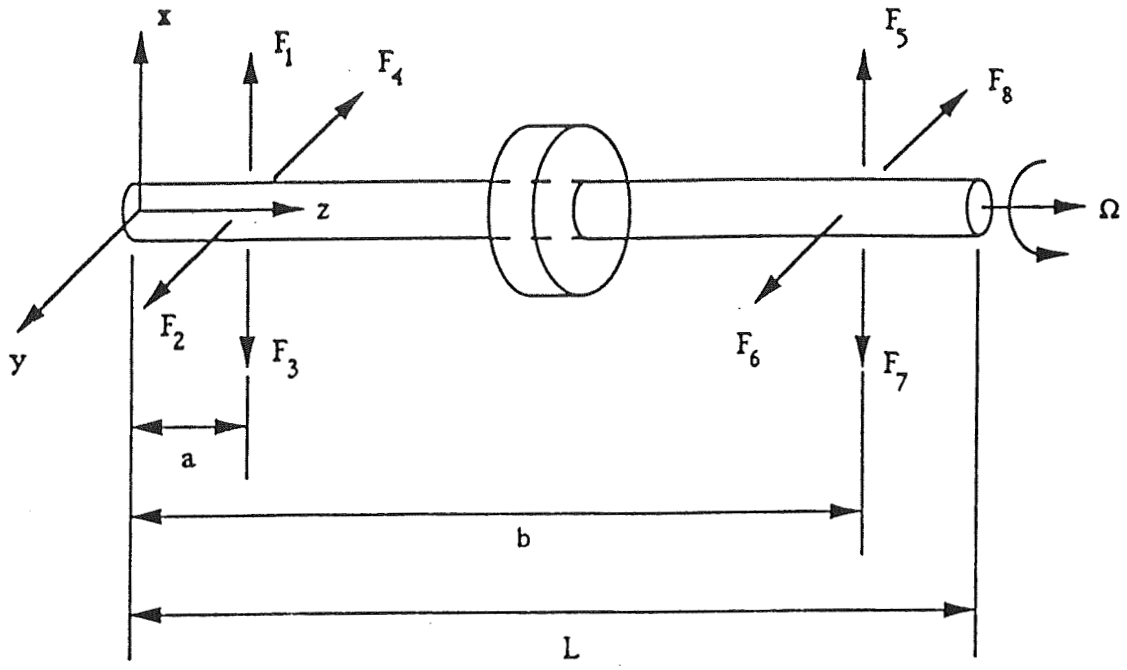


Figure 1: Generic Model of the Rotor Shaft

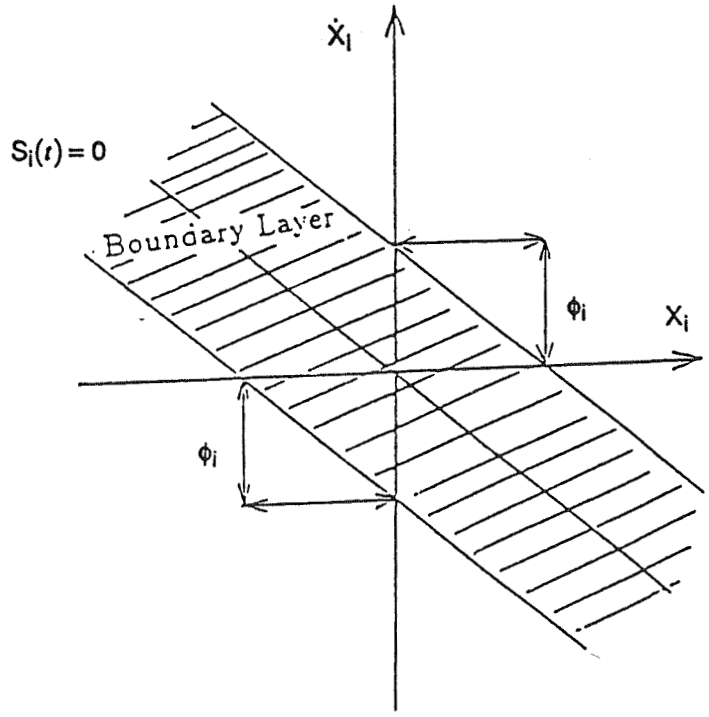


Figure 2: Illustration of Boundary Layer Concept

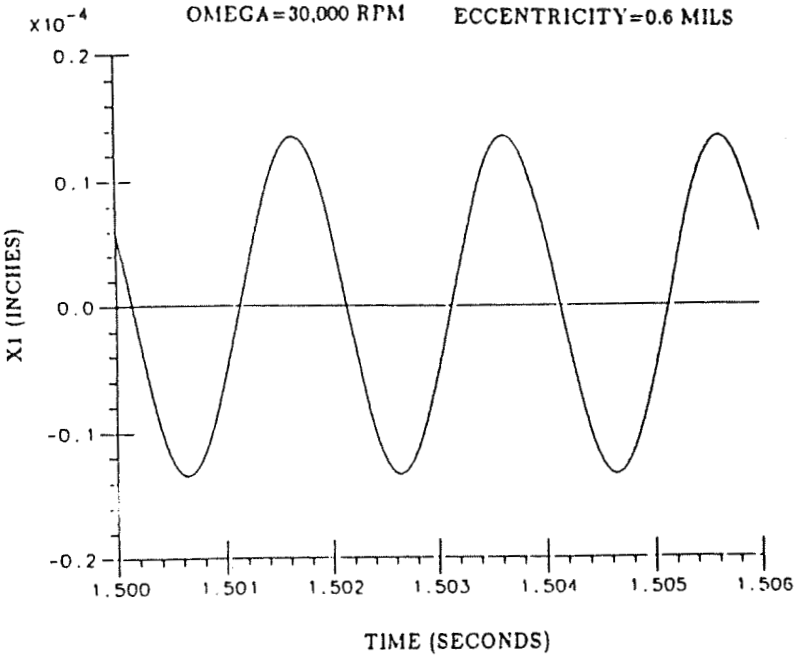


Figure 3: Steady State Response of the Controlled Rigid Rotor

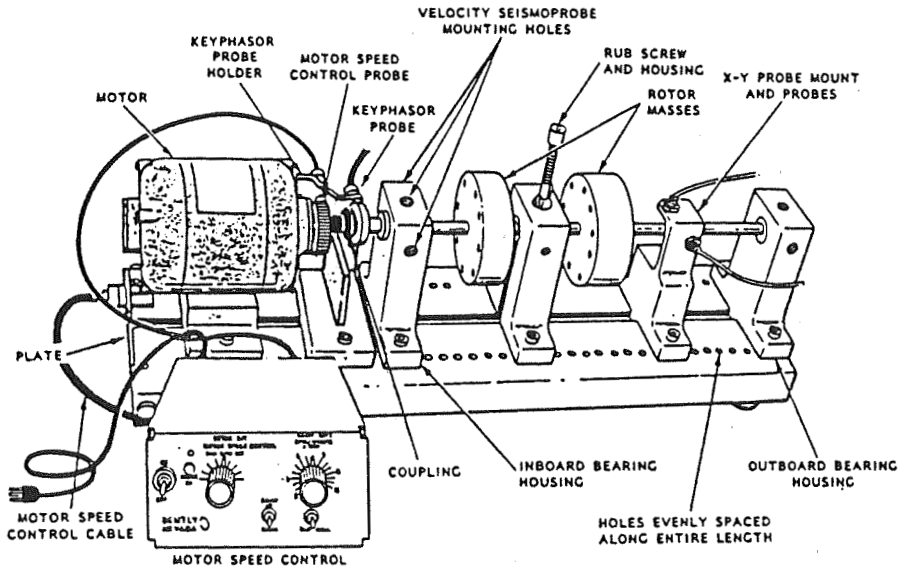


Figure 4: The Bentley-Nevada Rotor Fixture

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PRESENTATION 2.3.3

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ANALYSIS OF FOIL BEARINGS FOR HIGH SPEED OPERATION IN CRYOGENIC APPLICATIONS

ANALYSIS OF FOIL BEARINGS FOR HIGH SPEED OPERATION IN CRYOGENIC APPLICATIONS

Marc Carpino
Department of Mechanical Engineering

I. Research Objective and Potential Impact on Propulsion

The general objective of this project is to develop analysis tools which are required for the design of foil bearings to be used in cryogenic applications. During the second year of this project, a general analysis approach and code for journal bearings operating under steady state conditions will be completed. This will be followed by the initiation of an investigation into transient behavior of foil bearings to determine their performance in rotor systems.

Foil bearings have been proposed as an alternative to rolling element bearings for use in cryogenic turbo-pumps in liquid propellent rocket engines. This type of bearing offers several advantages over rolling element bearings since they would use the cryogenic pump fluid for a lubricant and have structural flexibility. These bearings have the potential of high reliability and long life.

The bearing surface is constructed of a "foil" which resists deflection by a combination of bending, membrane, and elastic foundation effects(see figs. 1 and 2).

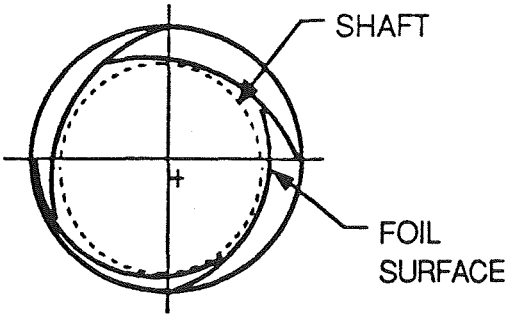


FIGURE 1 - LEAF TYPE BEARING

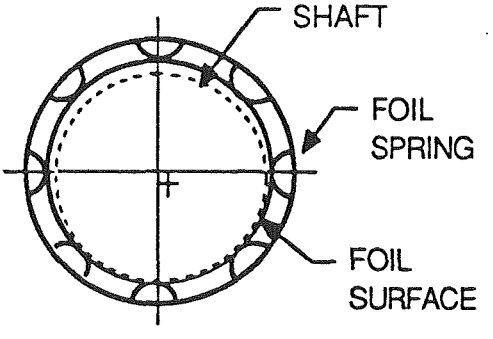


FIGURE 2 - HYDRESIL TYPE BEARING

The relative motion between the rotating shaft and the foil causes pressure in the fluid film to develop. This pressure deflects the foil surface away from the shaft. Once a full fluid film is established between the foil and the rotor shaft, contact no longer takes place and there is no subsequent wear of the surfaces. The flexible foil structure of the bearing allows it to compensate for minor tolerance and manufacturing defects. This same flexibility also has a significant effect on the dynamic performance of the rotor-bearing system.

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II. Current Status and Results

The initial efforts in this project have focussed on the development of a single unified approach which will address the broad range of possible configurations and operating conditions. A key part of the objective is to develop an analysis and code which is as modular as possible. The analysis process has been divided into two separate parts: determination of the nominal foil geometry for the bearing, and solution of the coupled foil deflection/fluid flow problem. After a discussion of this approach, results from a simple test configuration used to test the performance of solution approaches will be presented.

1. Calculation of Nominal Geometry

In a given design, the unconstrained foils can have arbitrary shapes and curvatures. A general nonlinear large displacement finite element formulation has been developed to calculate the nominal operating geometry of the foil. The formulation is used to calculate the nominal shape of the foil, the internal stresses in the foil, and the forces acting on the foil after the foil is assembled into the complete bearing. This information is then used to describe the foil in the coupled solution. The same deflection model, or a linearized version of it, can be used in the coupled solution.

2. Coupled Solution

The solution of the coupled problem requires the simultaneous calculation of the foil deflection, fluid flow, and the thermal transport. Within this project, a modified direct forward iteration approach has been developed for the solution of the coupled foil deflection/fluid flow problem. Direct forward iteration schemes normally utilize standard solution methods to iteratively calculate the displacement and the pressure. After each iteration, the deflection is updated and used to calculate a new pressure for the next iteration. The process is continued until the solution converges.

The iterative approach is modular and offers maximum flexibility in the development of alternate deflection and fluid models. The approach has been modified to incorporate the effects of the fluid flow, beyond the standard pressure coupling, into the finite element deflection calculation. This modification entails the addition of a stiffness matrix and loading vector based on the Reynolds equation to the deflection model. The Reynolds equation is still used to calculate the pressure in the fluid film from the clearance.

3. Results

The problem of an infinitely wide single foil bearing(see figure 3) has been used to test the modified direct forward iteration solution method. The foil is simply supported at the leading and trailing edges. A concentrated force is applied at the center of the foil. The slider, which is comparable in function to the shaft, is flat. This configuration is very similar to one of the foils which would be used in a multi-foil journal bearing configuration(see figure 4). In this case, the foil resists deflection by bending only.

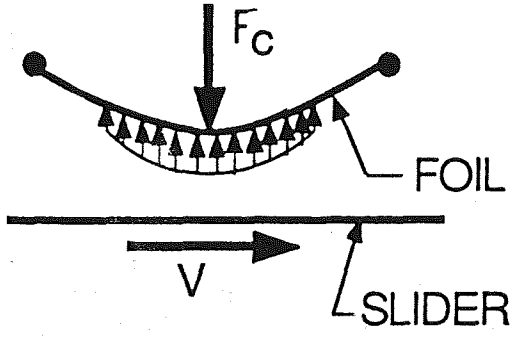


Figure 3 - Test configuration

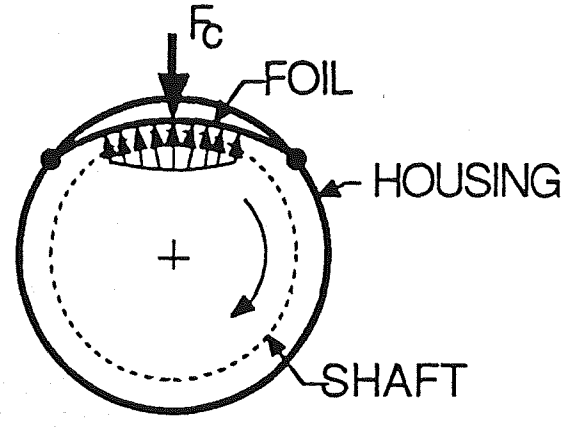


Figure 4 - Journal configuration

The significance of this new modified iterative approach is that it has significantly improved convergence characteristics compared to the standard direct iteration methods. This is of particular importance in heavily loaded bearings where the nominal deflection of the foil is large compared to the thickness of the fluid film. In heavily loaded cases, it is necessary to severely under-relax the displacement solution with very small relaxation factors to get converged solutions by standard direct iteration. The standard direct iteration solutions then require a large number of iterations, and convergence must be tested very carefully. The effectiveness of the modified iterative method is shown in figure 5, where the modified iterative approach converged significantly faster than the standard approach.

The clearance and pressure results of the modified iterative method for this same case are shown in figure 6. These results demonstrate the large change in clearance which is possible with this method. The method is very effective in preventing the clearance from becoming negative during the iteration process.

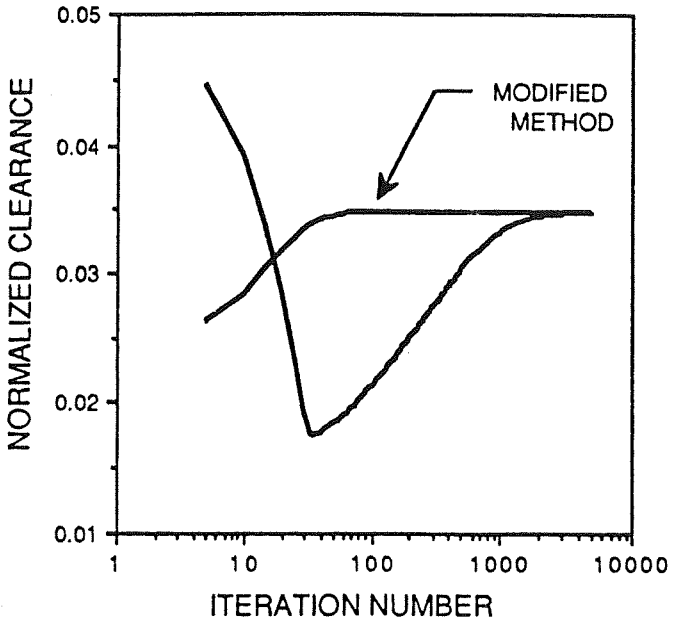


Figure 5 - Convergence of modified and standard iterative methods

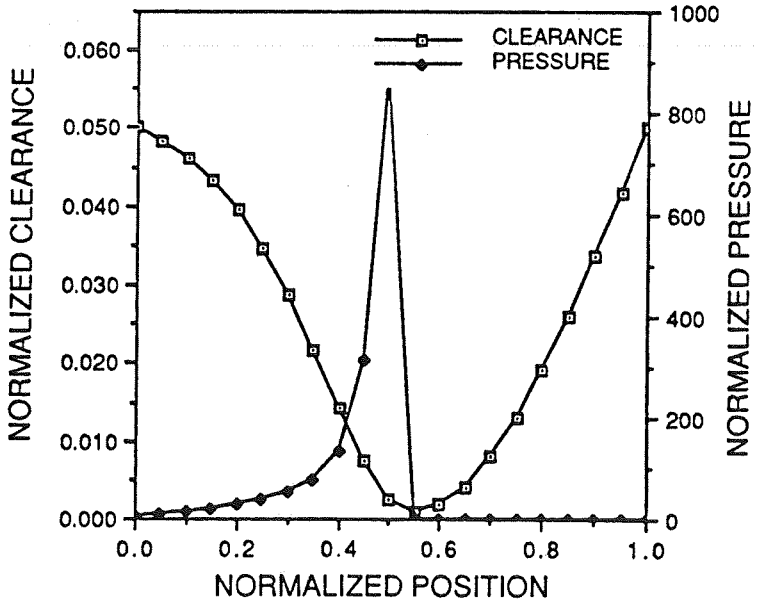


Figure 6 - Typical clearance and pressure distributions for heavily loaded bearing

III. Proposed Work for Coming Year

The efforts in the next year will be split between: the completion of the analysis and code for the steady state performance of foil journal bearings, and the start of an investigation into the dynamic behavior of rotor systems utilizing foil bearings.

Many of the critical issues in the steady state analysis have been addressed and resolved. The following tasks in the steady state analysis will be completed:

1. Complete the development of a second finite element code specifically for the solution of coupled problems. This code will use output from the nonlinear large displacement calculation to define the nominal foil geometry. It will utilize the same basic finite element formulation to solve the coupled problem by iteratively operating on different data sets for the deflection and fluid flow in the bearing. This will be a two dimensional code.

2. Implement the general analysis for a finite width, multiple foil configuration. Several element types and subroutines will be installed in the code described in the previous task. Simple configurations will be implemented first to allow checking of more complicated cases.

3. Investigate the significance of thermal effects. The behavior of the fluid cannot be investigated until a basic working model has been developed. Results of these calculations may lead to modifications of the element types.

The initial stage of the investigation of the transient behavior of these bearings will focus on the development of a suitable model which can predict bearing stiffnesses and damping for use in rotor dynamic calculations. The following issues will have to be resolved:

1. The significance of foil to foil contact and rubbing in determining the stiffness and damping of bearings. The interfaces between foils are not frictionless and may significantly impact the transient bearing performance by modifying the stiffness and providing frictional losses.

2. The relative contributions of the fluid film and foil rubbing to the stiffness and damping. Although the stiffness of the system may be controlled by deflection of the foils, changes in foil shape will affect the fluid film.

3. The interfaces between the foils most probably behave as coulomb friction contacts. An equivalent viscous model will have to be developed to approximate the performance in standard rotor dynamics models.

These investigations form the basis of an approach to the development of a transient model for the bearings to be used in rotor dynamics applications.

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PRESENTATION 2.3.4

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A STUDY OF METHODS TO INVESTIGATE NOZZLE BOUNDARY LAYER TRANSITION

A STUDY OF METHODS TO INVESTIGATE NOZZLE BOUNDARY LAYER TRANSITION

Laura L. Pauley
Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

Supersonic flow is a topic of strong interest which arises in applications ranging from the nozzle of a booster rocket to the National Aerospace Plane. Many characteristics of supersonic flow have been studied in detail and are well understood. The transition of a supersonic laminar boundary layer to turbulence, however, has been very difficult to study. Experimental measurements of transition are particularly difficult in rocket nozzle applications because the flow temperatures exceed 2300° C. In this case, only temperature measurements along the outside of the nozzle wall can be made. This yields the wall heat flux within the nozzle but does not indicate flow velocities or temperatures. It also does not provide a tool to predict the performance of nozzles.

To further investigate the nozzle flow, numerical computations are employed. The computations produce complete flow velocity and temperature fields within the nozzle. As a check these results can be compared with experimental data at the wall. Once an accurate numerical scheme has been validated, it can be used as a design tool to predict the performance of other nozzle designs without the cost of experimental testing. Typically the numerical analysis assumes either a laminar boundary layer or a fully turbulent boundary layer which is steady and two-dimensional. Boundary layer transition is not considered. Computing both the completely laminar boundary layer and the completely turbulent boundary layer conditions gives the minimum and maximum wall heat flux possible for a specified geometry. When the experimental heat flux measurements lie between these two values, the nature of the boundary layer is unknown. The boundary layer may have transitioned from laminar to turbulent, three-dimensional structures may be present in the boundary layer, or the inlet flow conditions may not be correctly specified in the computation.

In the NASA Lewis 1030:1 Area Ratio Nozzle, a series of experiments were conducted over a range of chamber pressures (Smith, 1988). The nozzle being tested was a low thrust nozzle design for space applications such as for orbital transfer vehicles. The throat diameter of the nozzle was 1 inch and the chamber pressure was varied from 360 to 1004 psi to give a thrust range of 500 to 1200 lbs. The heat flux measurements were compared with numerical predictions. At low chamber pressures, the experimental heat flux data corresponded closely to the laminar boundary layer computations. When higher chamber pressures were tested, the heat flux was found to be between the laminar and fully turbulent boundary layer predictions. The characteristics of the boundary layer were not correctly described by the laminar or turbulent calculation and therefore the heat flux was not predicted. Under these conditions, the nature of the boundary layer can not be inferred from the data.

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A boundary layer stability analysis can reveal the onset of transition or the growth of a three-dimensional structure in a laminar boundary layer. In the present study, a stability analysis will be used to investigate the supersonic boundary layer in a rocket nozzle. The study will focus on the NASA Lewis high-area-ratio nozzle conditions which experimentally produced wall heat flux values between those predicted by laminar and turbulent computations. Through a stability analysis, the location where transition begins and the structure of the most unstable disturbance can be predicted. Tollmien-Schlichting (planar waves) and Taylor-Görtler (longitudinal vortices) instabilities will be considered as possible transition mechanisms. This study will define the boundary layer region which is laminar and which can be predicted accurately by a two-dimensional, steady, laminar computation. Establishing the structure of the most amplified instability will then lead to an accurate model of the transition region. On the concave wall of the supersonic nozzle, it is expected that transition will be triggered by the Taylor-Görtler instability. The stability analysis will determine the wavelength of the disturbance which grows most rapidly. A three dimensional boundary layer computation will then predict the enhancement of the heat flux when a vortex structure of the dominant wavelength is present. These predictive models will be compared with the results from high-area-ratio nozzle experiments reported by Smith (1988). Once validated, the methods developed can be used in predicting the performance of new nozzle designs.

II. Current Status and Results

The research program will investigate the boundary layer structure found in high-area-ratio rocket nozzles. The experimental results of Smith, (1988) will be used to validate the numerical findings. All chamber pressures tested will be repeated in this numerical investigation. As higher chamber pressure results become available, numerical computations will also be conducted at those conditions. This research program will yield a predictive tool useful in analyzing other rocket nozzle designs. The research can be divided into three tasks, a boundary layer computation, a stability analysis and a transition model development.

The first task is to compute the laminar boundary layer flow throughout the entire length of the nozzle. This provides the mean flow which will be used in the stability analysis. At low chamber pressure conditions, the laminar computation should produce heat flux values similar to the experimental results. At high chamber pressures, the laminar heat flux predicted from the computations will be below the experimental values. This indicates that the experimental boundary layer begins to transition to turbulent flow. At the high chamber pressure conditions, a stability analysis will be used to predict the location where the laminar boundary layer begins to undergo transition. The first stage of the research has been started. A well-tested computer program is used to solve the compressible Navier-Stokes equations in the entire nozzle. The program uses a flux-splitting scheme and has been shown to give accurate results for a wide variety of nozzle geometries. Accurate results are also expected for the high-area-ratio nozzle of interest.

Accurate flow inputs and a smooth computational grid are required for accurate

numerical results. A chemical equilibrium and composition program* was used to determine the chemical composition and properties of the inflow gas using the experimental conditions cited by Smith (1988). A frozen flow (constant chemical composition) assumption was then made when computing the flow within the supersonic nozzle. Using tabulated properties for hydrogen-oxygen systems (Svehla, 1964), constants for the Sutherlands law viscosity expression were found so that the viscosity was accurately predicted throughout the entire temperature range of interest. A computational grid was produced which includes mesh clustering near the nozzle wall in order to resolve the boundary layer (see figure 1). Mesh clustering was also added near the throat where strong velocity gradients are expected. To minimize the skewness of the velocity vectors with respect to the grid cells, each spanwise grid line is a circular arc which is normal to the nozzle wall.

Presently we have obtained nozzle results when the chamber pressure is 360 psi. Within the diverging nozzle, the wall temperature was set to the average wall temperature measured experimentally. The velocity contours within the supersonic portion of the nozzle are shown in figure 2 and the Mach contours near the throat are shown in figure 3. In figure 4, the heat flux from the present computations is compared with the experimental results. The computational results are approximately 10% above the experimental results. For the low chamber pressure conditions, the computational results should accurately predict the experimental measurements.

To get a better comparison between the computational and experimental results, several modifications are currently being tested. The accuracy of the parameters in the viscosity model is being reexamined and improvements in the model across the operating temperature range will be implemented. The program is being modified so that a temperature distribution can be specified along the wall. Currently, the wall temperature is set to a constant in the entire supersonic region. The grid resolution near the nozzle wall and in the inviscid flow region is also being tested to assure that the numerical solution is grid independent. The modifications will lead to a more accurate representation of the problem and should also yield more accurate heat flux results. When the experimental heat flux results are correctly predicted, it will be inferred that the boundary layer flow within the nozzle has been correctly described by the computation.

After producing accurate results at the low chamber pressure, high chamber conditions will be computed. The laminar boundary layer results at the high chamber pressures will be used as the mean velocity profiles for the stability analysis. The location where the boundary layer begins to transition and the structure of the disturbance which triggers transition will be determined by the stability analysis.

III. Proposed Work for Coming Year

In the second phase of this research, the stability of the laminar boundary layer will be tested numerically. The stability analysis will indicate the structure and wavelength of the disturbance which will be amplified most rapidly and thus which will

* Referred to as CEC76 and developed by S. Gordon and B. J. McBride at the NASA Lewis Research Center.

cause the laminar boundary layer to undergo transition to turbulence. At every streamwise location, the amplification rate (α) for all wavenumbers is determined for both Tollmien-Schlichting and Taylor-Görtler instabilities. The amplification ratio (a) can be used to determine which disturbance has grown most rapidly throughout the boundary layer development. It is defined as the ratio of the amplitude of a disturbance to its amplitude at neutral stability.

$$a = \exp\left(-\int_{x_n}^{x} \alpha dx^{*}\right)$$

The disturbance which has the largest amplification ratio will initiate transition of the laminar boundary layer. Typically transition occurs when the amplitude ratio reaches e^9 or e^{10}. The criterion has been found to give an accurate prediction of the transition point when either Tollmien-Schlichting or Taylor-Görtler instabilities cause transition of compressible or incompressible boundary layers. This method for predicting the transition location is known as the e^N method.

Chen, *et al.* (1985) investigated the transition of the boundary layer in the diverging nozzle of a supersonic wind tunnel. Tollmien-Schlichting and Taylor-Görtler instabilities were considered along the curved wall. From a boundary layer stability analysis, they found that Taylor-Görtler instabilities grew more rapidly in the supersonic nozzle and caused the transition of the laminar boundary layer on the wind tunnel walls. Transition in the experimental facility occurred at the location where the amplification ratio from stability analysis had a value of e^9 to e^{11}. They suggest an amplification ratio of $e^{9.2}$ as a design criterion for transition.

To predict the location where transition begins, the e^N method will be used in the current investigation. The e^N method was successfully used to predict transition in the supersonic wind tunnel (Chen, *et al.*) and the method is expected to give accurate results in the current rocket nozzle study since the two applications have similar geometries.

Chen, *et al.* determined that the Taylor-Görtler vortices were responsible for the boundary layer transition in the wind tunnel nozzle. It is expected that Taylor-Görtler vortices will also trigger transition in the current investigation. The third task in the proposed research will be to predict the enhancement of the heat transfer due to the longitudinal vortices. To do this, a vortex array will be added to the boundary layer inflow of a three-dimensional compressible boundary layer computation. The wall heat flux will be determined and comparisons will be made with experimental results.

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Smith, T.A., (1988) "Boundary Layer Development as a Function of Chamber Pressure in the NASA Lewis 1030:1 Area Ratio Rocket Nozzle", NASA TM 100917. Also AIAA Paper 88-3301.

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Figures

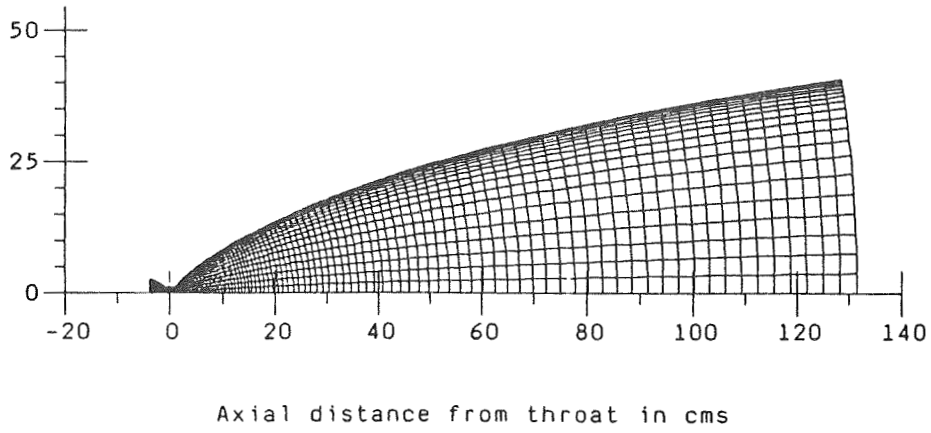


Figure 1. Computational grid.

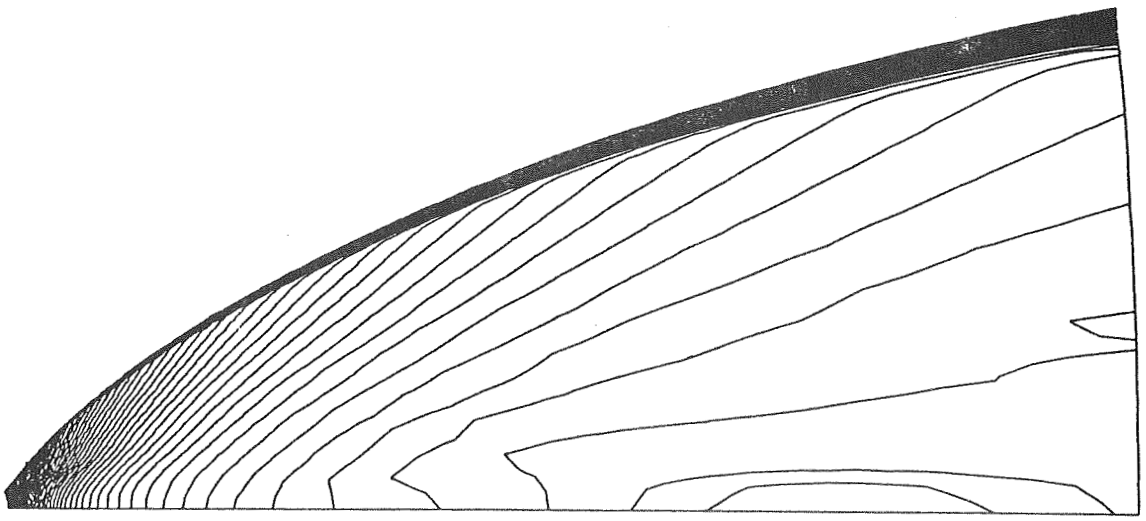


Figure 2. Velocity contours from computation.

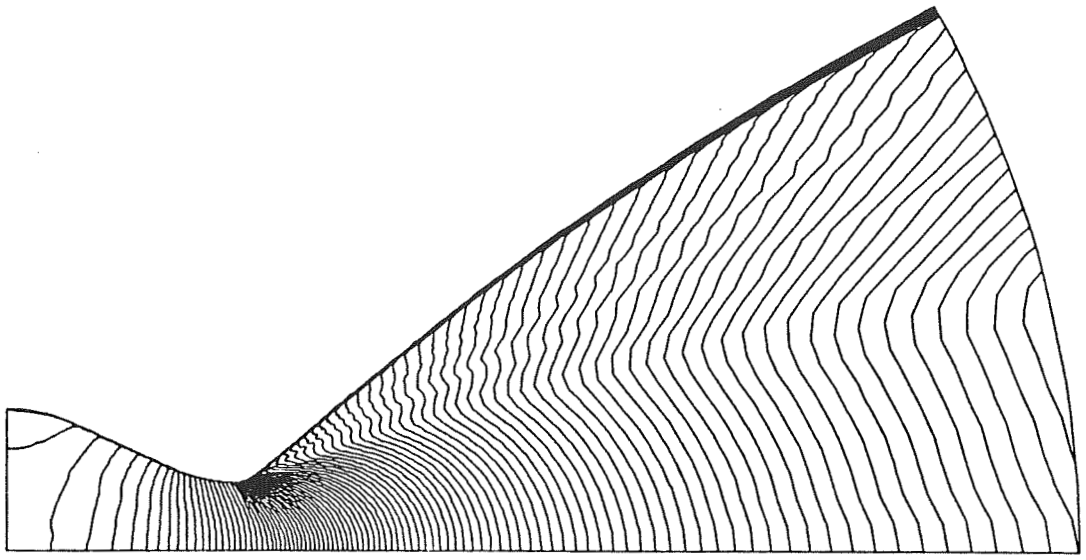


Figure 3. Mach contours near throat from computation.

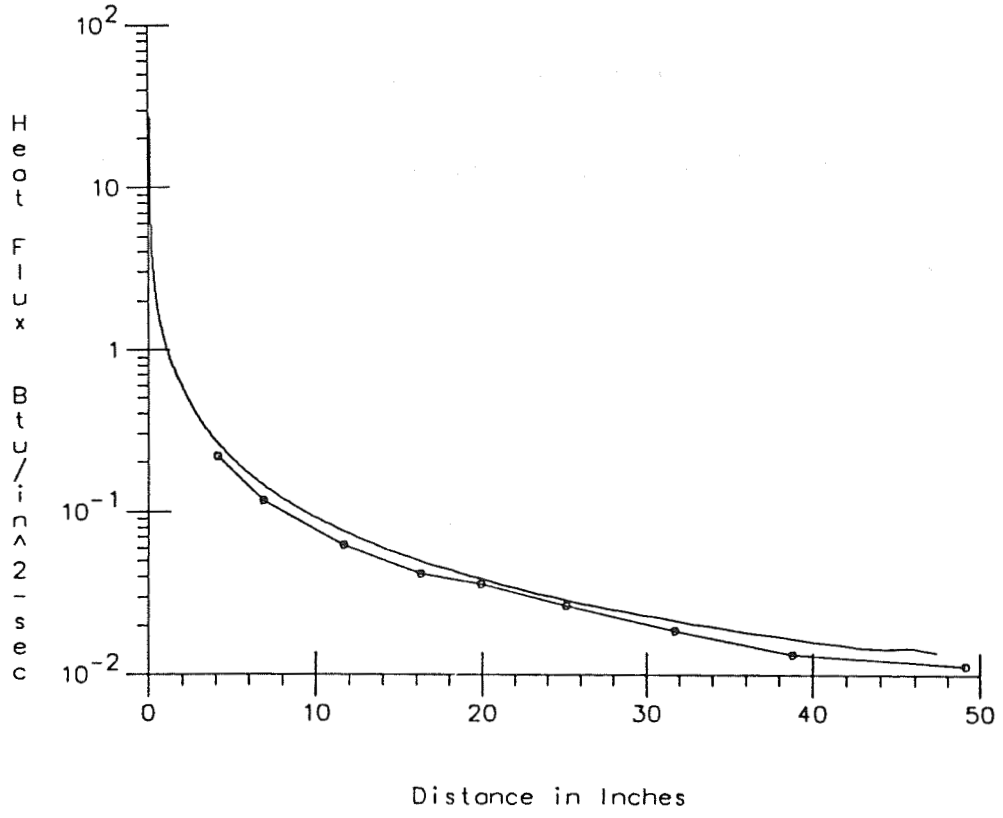


Figure 4. Experimental and predicted wall heat flux at a chamber pressure of 360 psia. (Smith, 1988)

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N91-28233

OPTICAL DIAGNOSTIC INVESTIGATION OF LOW REYNOLDS NUMBER NOZZLE FLOWS

OPTICAL DIAGNOSTIC INVESTIGATION OF LOW REYNOLDS NUMBER NOZZLE FLOWS

Michael M. Micci
Department of Aerospace Engineering

I. Research Objectives and Potential Impact on Propulsion

Small high performance chemical rocket motors and electric propulsion devices operating at low power levels (<25 kW) suffer from high nozzle losses due to the low Reynolds number of the exhaust flow. The Reynolds number with the diameter of the nozzle throat as the characteristic dimension is:

$$Re = \frac{\rho u D}{\mu}$$

$$\propto \frac{D P_o}{T_o}$$

As chemical and electric thruster research and development strives for higher chamber temperatures to increase performance, the future trend for nozzles will be for decreasing Reynolds numbers. Current methods for estimating rocket nozzle performance assume a boundary layer along the interior wall with an inviscid core. However at low Reynolds numbers the wall boundary layer fills most if not all of the nozzle interior producing a large amount of viscous dissipation. Nozzle efficiency, the ratio of the delivered specific impulse to the ideal specific impulse, has been measured as low as 60% at a Reynolds number of 200. Computational techniques are beginning to be able to examine such flows but the need exists for experimental verification of both numerical results as well as design changes in existing hardware.

The majority of experimental measurements to date have consisted of thrust and discharge coefficient, both global quantities which give little information about the detailed physical processes occurring. Different studies looking for the optimal nozzle wall contour for low Reynolds number flow among cones of various angles, bell-shaped nozzles and trumpet-shaped nozzles, reached different conclusions. The only detailed measurements of flow properties were taken by Rothe using an electron beam diagnostic in a low pressure nozzle flow. Rothe measured density and temperature as a function of axial and radial location in the nozzle and found that for low flow Reynolds numbers the static temperature would rise in the nozzle expansion due to viscous dissipation.

The objectives of this program are to obtain temperature, density and velocity profile measurements in the expansion region of low Reynolds number nozzles through the use of optical diagnostics. An LIF system will be used to probe the expansion of a microwave-heated expansion in the Center vacuum facility. The experimental measurements made in this program will be compared to numerical predictions obtained by Drs. Charles Merkle and Lyle Long.

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II. Current Status and Results

The majority of the effort expended to date in this program has been involved in the build-up of the Center vacuum facility. The vacuum chamber has a 1 meter inside diameter and is 5 feet long. Figure 1 is a diagram of the vacuum chamber with pump, microwave gas heater and LIF system. The Stokes mechanical pump and the Stokes diffusion pump were installed and connected with their requisites utilities, electricity, cooling water and exhaust vents, this past year. To date, the pumps have achieved a minimum pressure of $1.9 \cdot 10^{-4}$ Torr with no mass flow through the system. This corresponds to an altitude of 101 km. The diffusion pump has a zero flow rate specification of 10^{-4} Torr. Small leaks through the mounting threads of the two pressure transducers prevented a lower pressure from being achieved. Since both these pressure transducers were in temporary positions above the diffusion pump and are being relocated in the vacuum chamber, an extensive effort to seal these threads was not undertaken. However, new fittings for these transducers which will provide better high vacuum sealing are being procured.

With the check-out of the diffusion pump completed, the vacuum chamber has been mated with the diffusion pump.

A tunable laser system for use as a pump source for an LIF system has been identified. A Quanta-Ray pulsed Nd:YAG laser will be frequency doubled and used as the pump for a Quanta-Ray pulsed dye laser. This will provide an output at wavelengths of 654-647 nm for use in pumping the First Positive system of N_2. The output of the dye laser will be frequency doubled again to reach the 337 nm wavelength to excite the Second Positive system of N_2. Nitrogen has been chosen because its molecular nature will allow the examination of nozzle frozen flow losses due to nonequilibrium effects.

The LIF system will be used to obtain simultaneous measurements of temperature, density and velocity in the expansion flows from low Reynolds number nozzles. Density will be obtained from the absolute intensity integrated over the entire fluorescing line. An Optronics calibrated tungsten strip lamp has been procured and has been used successfully for emission spectroscopy absolute continuum measurements of the electron temperature in a microwave-heated helium plasma. Temperature of the heavy particles (neutrals and ions) will be obtained by measuring the Doppler broadened linewidth of fluorescing lines. Figure 2 shows the amount of Doppler broadening (including Stark broadening) for the Second Positive system of N_2 as a function of temperature. An electronically tunable Fabry-Perot etalon has been procured this past year which when combined with the Spex 0.5 meter monochromator which was purchased with funding from a prior grant should give a frequency resolution of 0.004 Angstroms.

The velocity of the gas will be determined from the Doppler shift of the fluorescing lines. Figure 3 plots the Doppler line shift of the 337 nm line of the Second Positive system for N_2 as a function of velocity. It can be seen by comparing Figures 2 and 3 that the Doppler line shift is much greater than the Doppler broadening, thus the broadening will not interfere with the velocity measurement.

Since the laser system is to be procured during the next fiscal year, the Fabry-Perot etalon is initially being used to measure the Doppler broadening and line shift of emitting lines to determine the temperature and velocity of a microwave-heated plasma expanding to atmospheric pressure and a velocity of Mach 1. Figure 4 is a drawing of the experimental system being used to make these measurements. The emitted light from the exhaust plume will be collected in the direction of the velocity measurement and compared to light collected in a direction where the gas has no directed velocity to determine the Doppler shift. The Fabry-Perot etalon is used as a very narrow bandwidth optical filter.

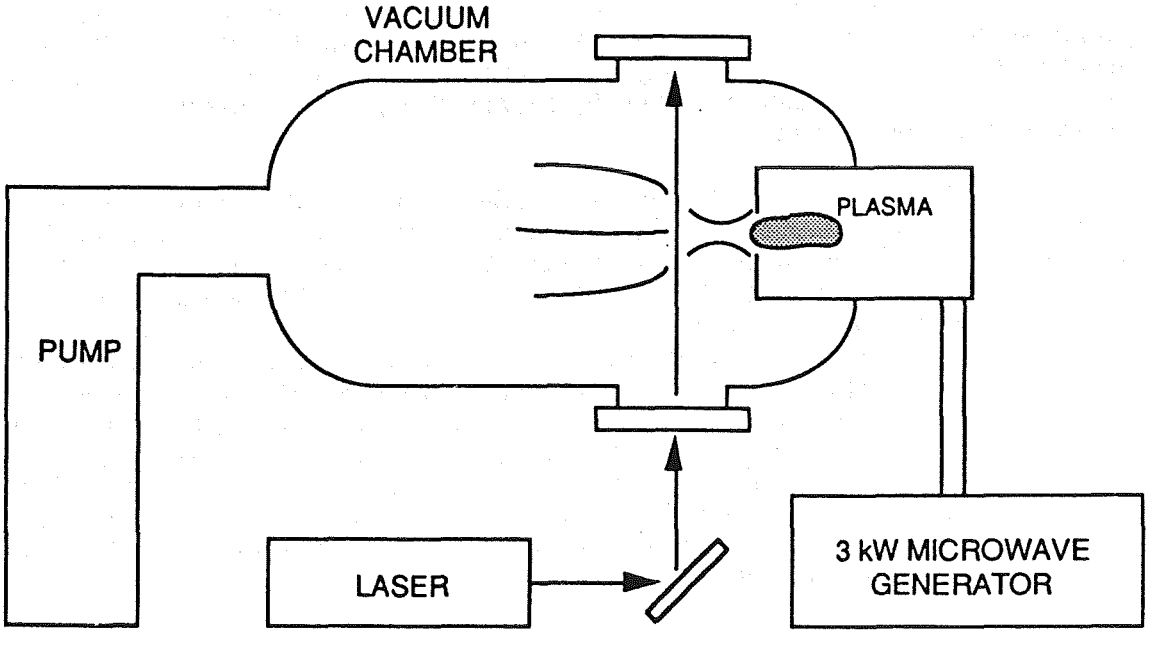


Fig. 1 Diagram of experimental facility to optically examine nozzle flows.

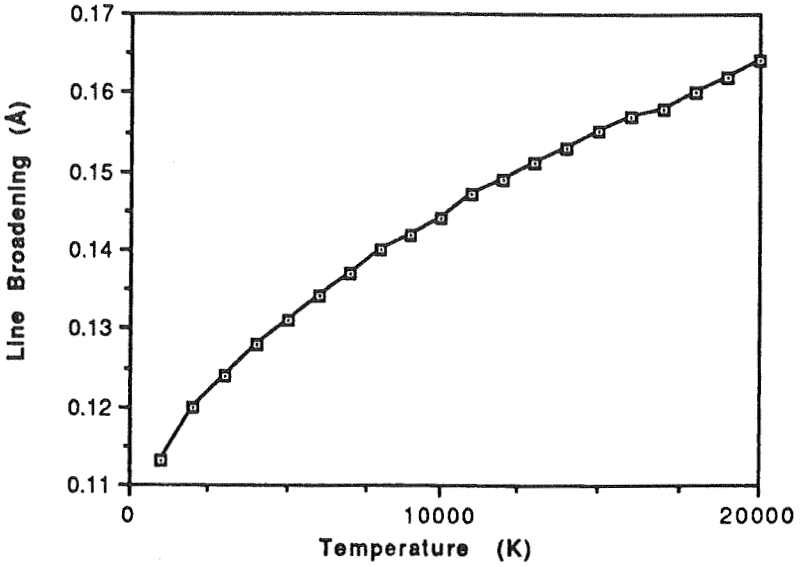


Fig. 2 Doppler line broadening for the 337 nm line of N_2 as a function of temperature.

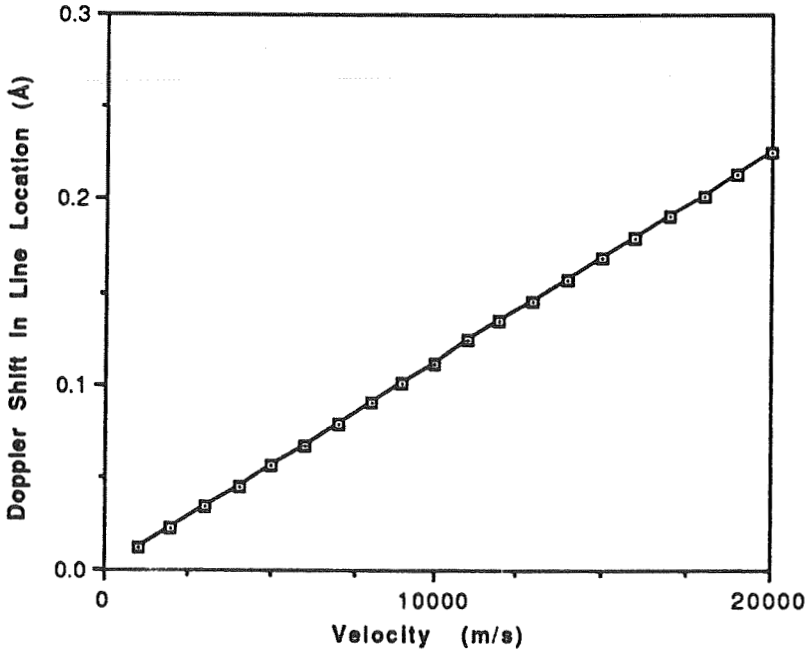


Fig. 3 Doppler line shift for the 337 nm line of N_2 as a function of velocity.

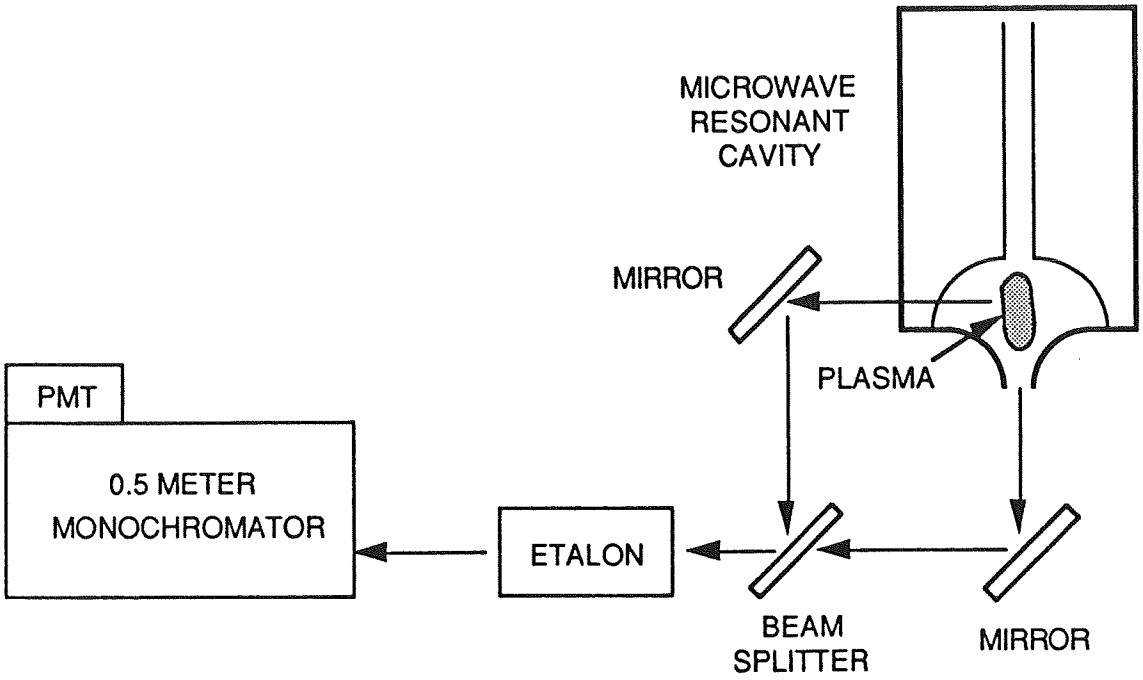


Fig. 4 Experiment to use Fabry-Perot etalon to measure Doppler line broadening and Doppler line shift of emitting microwave-heated plasma.

III. Proposed Work for Coming Year

The Stokes mechanical and diffusion pumps will be operated with the vacuum chamber attached and the minimum obtainable pressure as a function of mass flow rate and gas composition into the chamber will be measured. Early success with no gas flow through the system indicates that a good vacuum should be obtainable with experimentally significant mass flows. A fitting for the high vacuum pressure transducer which seals by the use of two O-rings instead of thread sealant is being procured and should solve the leakage problems experienced during the pump testing.

The interim tunable laser system being shared with Dr. Santoro for the LIF diagnostic should be in place in an adjacent laboratory by July 1, 1990. A system will be designed and constructed to divert the output beam through an opening in the wall between the laboratories into the window of the vacuum chamber which is located approximately nine feet above the laboratory floor. The laser beam will be intentionally reflected just prior to entering the window in order to aim it at the location of interest in the chamber. Additional windows are located on the chamber at positions opposite and perpendicular to the window through which the beam is entering while the nozzle exhaust is directed along the axis of the chamber.

The operation of the electronically tunable Fabry-Perot etalon as a very narrow bandwidth optical filter to measure Doppler broadened linewidths and Doppler shifts will be tested in an emission spectroscopy system. Gas will be heated to temperatures between 10,000 and 13,000 K in a microwave resonant cavity and accelerated to sonic velocities by expanding through a converging nozzle to atmospheric pressure. The principle disadvantage of emission spectroscopy compared to LIF is that emission spectroscopy is a line of sight measurement whereas LIF is a point measurement. Thus the initial results with the Fabry-Perot etalon will be somewhat degraded because of the averaging taking place through the measuring volume.

A circular microwave waveguide capable of applying the full 2.5 kW which is available to heat a gas is scheduled for delivery early in the grant year. A rectangular waveguide was used in prior experiments, however a rectangular waveguide generates a plasma which is attached at two locations to the waveguide walls. In order to generate a plasma which is not in contact with the waveguide walls, a circular waveguide was designed which will generate a plasma which is located along the waveguide axis. It will initially be tested with a choked converging nozzle expansion to atmospheric pressure and then integrated with the end flange of the vacuum chamber for expansions through converging-diverging nozzles to low pressures.

Finally, the use of LIF to measure density, temperature and velocity in the supersonic exhaust region of a microwave-heated gas flow will be evaluated. The laser will be used to excite the gas and the Fabry-Perot etalon in conjunction with the 0.5 meter monochromator will be used to measure the florescence signal. Spatial profiles of density, temperature and velocity in the radial direction will be measured. Initial nozzle geometries will be conical and property variations in the axial direction will be obtained by testing with nozzles of varying expansion ratios. Eventually, different nozzle contours such as bell and trumpet shaped will be tested.

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CFD APPLICATIONS IN CHEMICAL PROPULSION ENGINES

CFD APPLICATIONS IN CHEMICAL PROPULSION ENGINES

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Department of Mechanical Engineering

I. Research Objectives and Potential Impact on Propulsion

The present research is aimed at developing analytical procedures for predicting the performance and stability characteristics of chemical propulsion engines. Specific emphasis is being placed on understanding the physical and chemical processes in the small engines that are used for applications such as spacecraft attitude control and drag make-up. The small thrust sizes of these engines lead to low nozzle Reynolds numbers with thick boundary layers which may even meet at the nozzle centerline. For this reason, the classical high Reynolds number procedures that are commonly used in the industry are inaccurate and of questionable utility for design. A complete analysis capability for the combined viscous and inviscid regions as well as for the subsonic, transonic and supersonic portions of the flowfield is necessary to estimate performance levels and to enable trade-off studies during design procedures.

Most engines that are used for auxiliary propulsion operate at efficiencies that are considerably below those reached by larger engines. Although a portion of this efficiency decrement is due to the reduced Reynolds number conditions, a substantial fraction can also be attributed to the design processes which fail to take into account properly the viscous nature of the flow. Improved design and analysis capabilities should allow considerable performance improvements in these small engines. Higher performance in turn means that a reduced amount of propellant is needed to keep the spacecraft on orbit, thus leading to longer spacecraft life without increased launch weight. This potential for increased on-orbit time is the justification for the present research effort. Additional areas that are being considered include CFD modeling of combustion instability. These efforts are directed, for the most part at larger engines, but again, the application of CFD here promises to provide increased understanding of the physics that control engine design.

The numerical analyses of compressible flows has progressed rapidly in the past two decades,and it is presently routine to compute two-dimensional steady flows, while two-dimensional unsteady and three-dimensional flows are feasible in a research environment. Although two-dimensional computations are within the reach of day-to-day design procedures, no particular attention has been given to the computation of rocket flowfields and care must be taken to develop a procedure that is accurate and efficient enough for design use. Similarly, the areas of combustion instability is one that has received little attention from full CFD formulations. In addition to the computational aspects, major modeling issues exist with regard to the spray atomization and vaporization processes as well as the turbulent combustion processes and the heat release rates in the combustor. Three-dimensional and unsteady flows are also of downstream interest. The present effort is directed towards these issues.

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II. Current Status and Results

At present, coordinated efforts are going on in several areas. The first has to do with the application of CFD methods to the prediction of mixing and combustion in auxiliary propulsion engines . Our efforts here have started from an existing code for hypersonic reacting flows. This code uses an LU-SSOR central-difference algorithm with a complete chemistry and physical properties formulation for hydrogen-oxygen reactions and kinetics. Turbulence is modeled by the simple Baldwin-Lomax mixing length model. Although this procedure is effective for hypersonic flows, it has some shortcomings in the low-speed subsonic regime that is characteristic of the heat release regions in chemical propulsion engines. Several enhancements are being investigated to increase the robustness and efficiency of the code in this lower speed regime.

The first enhancement in the code was to switch from the LU solution algorithm to a fourth-order, explicit Runge-Kutta procedure with an implicit treatment for the axisymmetric and chemical kinetics source terms. The implicit treatment of the source terms enables larger time steps in applications where the source terms are stiff. All terms are still centrally differenced on a finite volume grid in a manner identical to the original LU formulation. Some representative results comparing the new RK4 procedure with the LU method are presented on Fig. 1. These results show that the Runge-Kutta method converges approximately twice as fast as the LU procedure in terms of iterations, in addition to showing faster CPU times per iteration. The LU procedure requires about 135 μsec. per grid point per iteration on the CRAY-YMP, while the RK method requires only 98 μsec.

Other enhancements to the code include the implementation of characteristics-based boundary conditions on the inflow and outflow boundaries (which also enhance the convergence of the original LU method as Fig. 1 shows). In addition to implementing characteristic techniques, boundary procedures were also added to enable the specification of the incoming mass flow rather than only stagnation quantities. This enhancement enables the user to mimic the experimental procedure where upstream conditions control the incoming flow rates of fuel and oxidizer while the choked throat in the nozzle sets the chamber pressure. Experience to date with the RK method shows that it is more robust than the LU method for this problem, and, in addition, requires smaller amounts of artificial viscosity to be added, thus enhancing the accuracy of the results, particularly in the steep gradient regions near the wall.

Some representative solutions for a gaseous hydrogen-oxygen engine are shown on Figs. 2 to 4. Figure 2 shows the overall geometry of the proposed station-keeping engine for the Space Station, while Fig. 3 shows the computed temperature contours and Fig. 4 the corresponding Mach number contours in the downstream mixing region of the low Reynolds number combustor. The computational grid used for these results is given in Fig. 5. The results show that the hydrogen stream from the outer periphery of the nozzle undergoes little reaction with the internal oxygen-rich core stream. Although these predictions could be correct, the present turbulence model is not sufficient to predict accurately the mixing and combustion processes in

this complex boundary-layer/free-shear-layer region. For this reason we are currently adding a k-ε turbulence model to the code to enhance these predictions.

At the present time the k-ε model has been coded and is being debugged and validated. The model is presently working for the boundary layer in the outer hydrogen stream alone, and current efforts are directed toward demonstrating it for the shear layer region as well. The model is formulated with low Reynolds number terms to enable the profiles to be computed all the way to the wall. Preliminary results suggest that this more complex turbulence model will predict substantially faster mixing and combustion processes than the present mixing length model, although comparison of the code predictions with hydrogen-oxygen flame measurements obtained from the literature is necessary to verify its accuracy. Downstream plans include extension to a complete PDF model of the turbulent combustion process.

Additional efforts on the low Reynolds number nozzle problem include the implementation of a flux-difference-split, upwind-biased, finite volume method with TVD capabilities for the flowfield. Results to date show that the upwind-finite volume method is considerably more robust than the centrally differenced methods, and should result in a more reliable code. The use of third-order biased differencing provides accuracy that is similar to that obtained with central differences. All results to date are for air chemistry, and again for the Baldwin-Lomax turbulence model, and are based on an alternating direction implicit (ADI) solution procedure. Modifications for the chemistry and physical properties of hydrogen-oxygen mixtures is nearly complete and results should be available shortly. The incorporation of an LU time marching procedure, that is also underway, should provide a further improvement in computational efficiency over the RK4 method. Although the LU procedure is not efficient for centrally differenced schemes, it is quite effective for upwind schemes, and substantial improvements in speed are expected.

A third effort in developing robust procedures for the small rocket engine application is adaptation of a parabolic Navier-Stokes (PNS) based algorithm to fully elliptic flow in the supersonic portion of the nozzle. This method has been demonstrated for perfect gas flows and is currently being extended to the full hydrogen-oxygen kinetics scheme, again using the same kinetics package that was used above. Our current efforts here are on the development of the space-marching PNS procedure. Later extensions will add a reverse iterative sweep to incorporate the complete elliptic Navier-Stokes effects. Some representative results for the perfect gas case in a contoured, high expansion nozzle, including the effects of expansion to a non-ideal back pressure are shown on Figs. 6 and 7. The PNS method, being a single pass method, is much faster than iterative methods, and the forward-backward sweeping procedure for the full Navier-Stokes equations is again much faster than classical time-marching algorithms.

Other efforts include the addition of a time derivative preconditioning method for enhancing convergence in low speed regimes such as occur in rocket combustors and the development of an Eulerian-Lagrangian method for oxidizer droplets. The convergence rates of most time-marching procedures scale inversely with the flow Mach number because of the stiff eigenvalues in the low speed regime. Because most rocket combustion occurs at these low Mach numbers, it is imperative that methods be

developed to retain fast convergence rates especially when complex chemistry models are in use. The present effort is based on extending earlier preconditioning methods by the author for inviscid flows to flows where diffusion is significant. Figure 8 compares the convergence rates with and without preconditioning for typical low speed conditions.

The work on two phase flows is in an early stage with efforts currently focussed on reviewing current state-of-the-art modeling procedures for liquid spray combustion. At present a first generation liquid propellant model has been developed and applied to one-dimensional flows. Our immediate plans are to extend this analysis to two dimensions by combining our existing Eulerian methods for the gas phase with Lagrangian droplet models developed elsewhere.

Finally as a last effort, a CFD-based combustion instability model is being developed for predicting finite amplitude waves in rocket combustors. At present, efforts are concentrated on developing accurate procedures and we are therefore employing simple combustion models (equivalent to n-τ models) with perfect gas conditions. Later extension to more realistic combustion models is planned. The model is based on an iterative, implicit time-marching method with capabilities similar to those described above for low speed flows. At present we are comparing one and two-dimensional oscillatory wave calculations with test cases based on linearized stability results that have been recommended by the current JANNAF Combustion Instability Panel as the result of recent JANNAF workshops.

III. Proposed Work for Coming Year

For the coming year our efforts are to be divided among several fronts. Primary emphasis is to be placed on enhancing and upgrading the turbulent combustion model and validating the code. Of secondary interest is a continuing effort to make the algorithm more robust and reliable both through day-to-day running, and through the addition of algorithm enhancements. Additional improvements are also planned to enable a wider, choice of wall boundary conditions and geometrical configurations. Low level efforts are also planned for the continued development of a liquid spray capability in the code. The final item to be addressed is the continued development of a combustion instability model based on CFD procedures.

The efforts on improved turbulent combustion models will focus on completing the incorporation of a two-equation model of turbulence to enable us to represent the boundary-layer-shear-layer more realistically. This model will then be augmented by a turbulent combustion model, most likely of the PDF variety using existing techniques from the literature. The validation of the code by comparison with experimental data will also be an area of focus. In conjunction with advanced turbulence modeling, we will also look for methods for enhancing the mixing and combustion processes in the rocket combustor. This will probably necessitate the use of three-dimensional phenomena, and, in particular, will include an assessment of the effect of discrete hole injection of hydrogen fuel (as is presently being done in the experimental configuration) on the mixing and combustion of the Space Station engine. These discrete holes are presently being modeled in two-dimensional fashion as a slot, and

this simplified treatment may be underestimating the degree of mixing and combustion. Improved methods for treating this three-dimensional injector pattern without going completely to three-dimensions will also be sought, including efforts for estimating the errors introduced by the use of the current two-dimensional predictions instead of the more realistic three-dimensional geometry.

In terms of a continued upgrade of the code, the results of other companion studies concerning various methods for enhancing the present algorithm will be integrated into the model as appropriate. Specific issues to be considered include the efficacy of switching to the flux-difference-split method, the enhancements to be gained through the use of low Mach number preconditioning, and the desirability of incorporating the PNS-based Navier-Stokes algorithm for the supersonic portion of the nozzle flowfield. The emphasis in these code improvement issues will be to evolve the code steadily into a more reliable procedure, as capabilities dictate. In addition, the extension of the method to other chamber geometries will also be undertaken, with the purpose of determining the capabilities of the method for predicting thrust and specific impulse accurately.

Additional capabilities to be included in the analysis include the addition of regenerative and radiation wall cooling procedures so the nozzle flowfield calculations will predict both the wall temperature and the heat flux distribution. The accurate prediction of wall heat transfer is an important issue in the design of small rocket engines, and requires an adequate model of turbulent wall heat transfer as well as some estimate of the locations of transition to turbulence. The improved turbulence model should provide some help in this area also.

Efforts to model the two-phase liquid spray combustion process will continue, although at a lower level of effort than the above topics. The purpose here will continue to be focussed on identifying the appropriate levels of technology and assessing the degree to which the spray combustion process can be predicted. Only representative solutions of simpler combustion processes are expected to be completed this year. A major emphasis will be placed on assessing the degree of reliability that can be expected as a stepping stone for planning the following year's effort.

In the related area of combustion instability, primary emphasis will be placed on demonstrating the accuracy that can be expected from a CFD analysis of combustion instability using a global model for the combustion process. Emphasis will be placed on estimating the effect of distributed heat release on disturbance growth, the effects of finite nozzle lengths of representative size, and other similar analyses based on simple phenomenological combustion models. These calculations will center around two-dimensional, radial-tangential modes, but additional studies of complete three-dimensional models based on the numerical solution of the linearized equations will be obtained as a precursor to incorporating three-dimensional disturbances in the complete nonlinear solution. Additional emphasis will be placed on the effect of finite amplitude disturbances on the growth of waves. An assessment of the advantages of CFD analyses over classical linear stability procedures will also be a goal of this research along with a comparison of the relative merits of the CFD predictions.

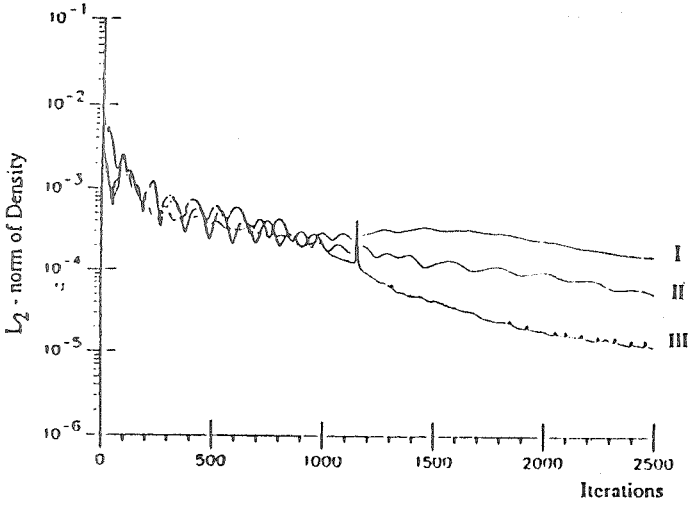


Figure 1. Convergence Histories: I) Original LU scheme (135 μs/grid point/iteration), II) Modified LU (123 μs/pt-iteration), III) Four-step Runge Kutta (98 μs/pt-iteration).

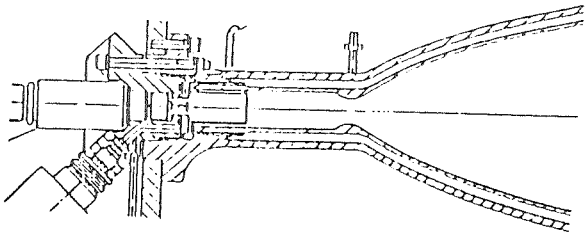


Figure 2. Aerojet Space Station Thruster #2

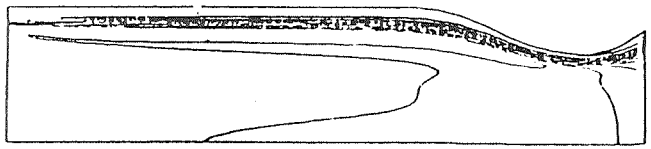


Figure 3. Temperature Contours

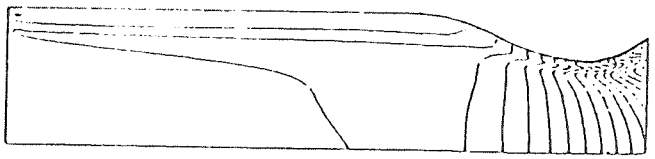


Figure 4. Mach Number Contours

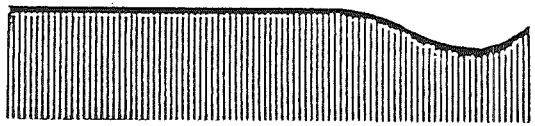


Figure 5. 90 x 60 Computational Grid

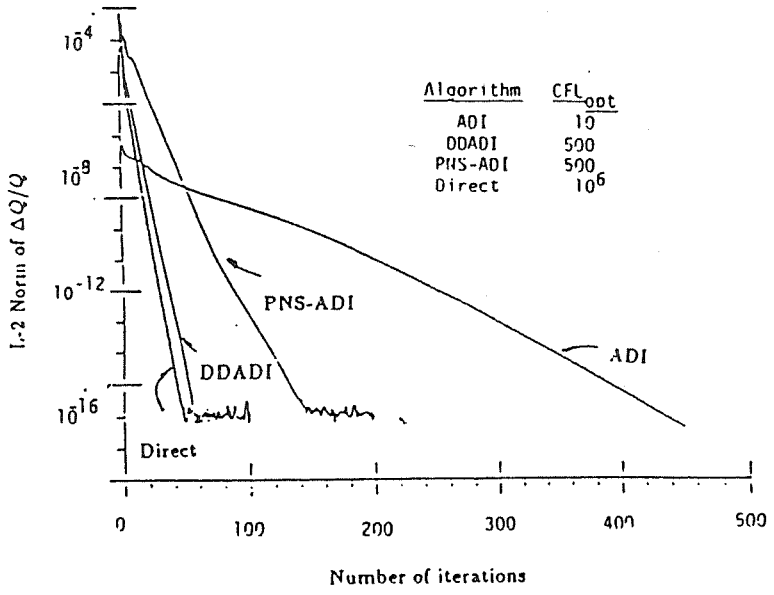


Figure 6. Comparison of Convergence for PNS-ADI Scheme with ADI Method for Supersonic Nozzle Flow.

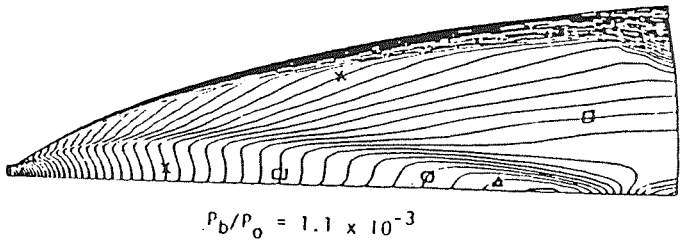


Figure 7. Mach Number Contours for Turbulent Flow in Overexpanded 272:1 Contoured Nozzle.

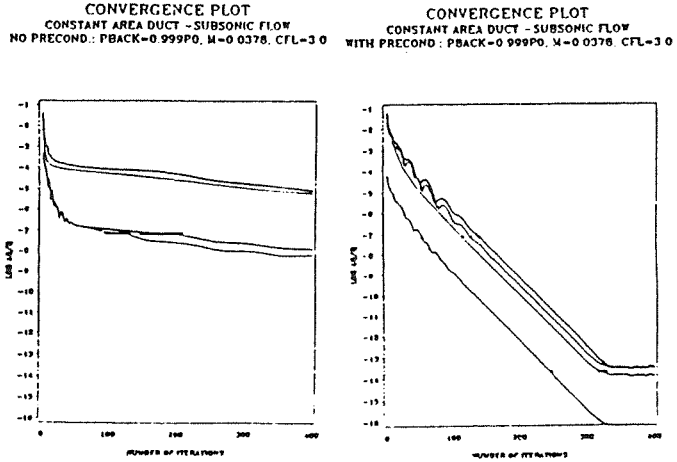


Figure 8. Effect of Time Derivative Preconditioning on Convergence in Low Mach Flows Representative of Combustor Conditions.

III. Proposed Work for Coming Year

1. Complete the development of the droplet temperature measurement technique and its integration with the droplet size and velocity techniques in order to obtain the capability to make simultaneous droplet size, velocity and temperature measurements in liquid hydrocarbon sprays.

2. Complete the fabrication, assembly and testing of the 70 atm, 300 • C turbulent flow system and single droplet generator.

3. Obtain a set of simultaneous droplet size and velocity measurements in a vaporizing liquid hydrocarbon spray, at atmospheric pressure and room temperature, and at one laminar and one turbulent flow condition.

4. Obtain a set of droplet size and temperature versus time measurements for the case of individual liquid hydrocarbon droplets injected into the same two flow conditions used in task 3.

5. Make a preliminary study of the behavior of individual liquid hydrocarbon droplets injected into a supercritical environment using high speed, back lit photography.

6. Make a preliminary investigation of the feasibility of using two-dimensional Raman scattering to visualize supercritical liquid hydrocarbon droplets.

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15. Supplementary Notes

16. Abstract

The Space Transportation Propulsion Technology Symposium was held at the Pennsylvania State University in State College, PA, June 25-29, 1990. The Symposium was held to provide a forum for communication within the propulsion technology developer and user communities. Emphasis was placed on propulsion requirements and initiatives to support current, next generation and future space transportation systems, with the primary objectives of discerning whether proposed designs truly meet future transportation needs and identifying possible technology gaps, overlaps and other programmatic deficiencies. Key space transportation propulsion issues were addressed through four panels with government, industry and academia membership. The panels focused on systems engineering and integration; development, manufacturing and certification; operational efficiency; and program development and cultural issues.

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